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AFFTC

# LIMITED HIGH ALTITUDE PERFORMANCE EVALUATION OF THE HH-53C HELICOPTER


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TECHNICAL REPORT No. 71-54

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Distribution limited to U.S. Government agencies only (Test and Evaluation), November 1971. Other requests for this document must be referred to ASD (SDQH), Wright-Patterson AFB, Ohio 45433.


AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE

DEPARTMENT OF THE AIR FORCE
HEADQUARTERS AERONAUTICAL SYSTEMS DIVISION (AFSC)
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

REPLY TO ATTN OF: ASD/SDQH 5-27 (Maj Thompson/54921/cal/H-53/R&D 9-2)

SUBJECT: ASD Addendum to FTC-TR-71-54, H-53 Altitude Performance

TO: Recipients of FTC-TR-71-54

This report is a part of and should remain attached to FTC-TR-71-54, "Limited High Altitude Performance Evaluation of the H-53C Helicopter". Paragraph numbers below correspond to the recommendations in the AFFTC Technical Report.

1. Concur. ASD will accomplish the data analysis required to incorporate this test data in the flight manual (along with data from previous tests, which is being reduced by commercial contract). All available flight test data will be incorporated simultaneously.

2. Concur with intent. ASD has initiated actions to incorporate the required information in the flight manual.

3. Concur. Test data analysis indicates that the largest increase in hover weight capability (at 9500 feet, standard day, and 100% $N_r$) due to the new bellcrank alone is approximately 2650 pounds. ASD provided technical approval of the Engineering Change Proposal (ECP 7379) with the recommendation that WRAMA procure the bellcrank for the H-53 fleet.

4. Concur. The ASD data effort for the flight manual will consider this compromise of optimum performance in favor of safety.

5. Concur with intent. ASD has established an additional data analysis task to determine the feasibility of defining single engine performance without further flight tests of H-53 helicopters. This effort is undertaken to determine the extent of additional test data which may or may not be required to further define the single engine height-velocity characteristics, if deemed necessary at a later date.

FOR THE COMMANDER

WILLIAM D. EASTMAN, JR., Lt Col, USAF
Chief, Helicopter Program Office
Directorate of Combat Systems
Deputy for Systems

PRIDE IN THE PAST — FAITH IN THE FUTURE

FTC-TR-71-54

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# FOREWORD

This report presents the results of the limited high altitude performance tests and tail rotor evaluation of an HH-53C helicopter, USAF serial number 68-10354. Testing was conducted between 26 September and 16 November 1971 at Bishop, California, and the nearby high altitude test site, Coyote Flats. The tests were conducted under authority of AFFTC Project Directive 72-17.

The authors of this report wish to express their appreciation to Airman Gary Snitily for his assistance with the data reduction and engineering analysis. In addition, the dedicated efforts of SMSgt Frank M. Presbury and his maintenance crew are most thankfully acknowledged.

Foreign announcement and dissemination by the Defense Documentation Center are not authorized because of technology restrictions of the U.S. Export Control Acts as implemented by AFR 400-10.

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# ABSTRACT

This limited flight test program defined the hover and takeoff performance of the HH-53C helicopter at high altitude and included an evaluation of the increased tail rotor power provided by a new tail rotor bellcrank. Hover and takeoff performance were satisfactory during high altitude testing. Operating at 105-percent rotor speed at high altitude, heavy gross weight combinations provided a more rapid response of the aircraft to control inputs than operation at 100-percent where control response was somewhat sluggish. In addition, it provided increased performance, prevented reaching mechanical control stops from limiting performance and lowered cruise guide readings. The new tail rotor bellcrank increased the HH-53C lift capability approximately 4,500 pounds by preventing tail rotor authority from limiting aircraft performance. A compromise between maximum performance and safety resulted in recommendation of a 35 knot indicated airspeed for takeoff.

# table of contents

# list of illustrations

# list of abbreviations and symbols

| Item | Definition | Unit |
|---|---|---|
| A | rotor disk area | $ft^2$ |
| AFCS | automatic flight control system | - - - |
| avg | average | - - - |
| C | centigrade or Celsius | - - - |
| $C_p$ | pressure coefficient | dimensionless |
| $C_T$ | thrust coefficient | dimensionless |
| cg | center of gravity | in. |
| dh/dt | time rate of change of altitude | ft per min |
| FAT | free air temperature | deg C |
| fwd | forward | - - - |
| IGE | in ground effect | - - - |
| KIAS | knots indicated airspeed corrected for instrument error | - - - |
| $N_r$ | rotor speed | rpm |
| Q | engine torque | pct |
| R | rotor radius | ft |
| R/C | rate of climb | ft per min |
| rpm | revolutions per minute | - - - |
| SHP | shaft horsepower | $550 \frac{\text{ft-lb}}{\text{sec}}$ |
| $T_a$ | ambient temperature | deg K |
| W | airplane gross weight | lb |
| $\rho$ | air density | slugs per $ft^3$ |
| $\Omega$ | rotor angular velocity | rad per sec |

# INTRODUCTION

Previous HH-53C hover performance testing as reported in references 1 and 3 was accomplished only at sea level. As a result of the earlier tests, it was recommended that high altitude testing be accomplished to adequately define the operational envelope of the aircraft (reference 1). In addition, the HH-53C has experienced growth in its performance capabilities. Engine power has increased from an original 3,435 shaft horsepower (SHP) to the present value of 3,925 SHP, the main gearbox's torque capability has increase 18 percent, and for the tests reported herein, $T_5$ limits were raised from 727 to 750 degrees C. These increased performance capabilities resulted in a tail rotor control problem as was reported in reference 4. Reports from operational users noted that the tail rotor does not have sufficient control power to overcome the torque generated by the main rotor at high altitude and heavy gross weight combinations. To prevent lack of tail rotor control from limiting the performance of the HH-53C a new tail rotor bellcrank was installed on the test aircraft which increased the available tail rotor blade angle by 3.25 degrees. Effects of the new tail rotor bellcrank were evaluated during these high altitude tests. Also, limited takeoff performance, sawtooth climbs, and a qualitative approach evaluation were accomplished on an opportunity basis.

All tests were conducted with engine air particle separators (EAPS), 450-gallon tip tanks and rescue hoist installed. Thirty-two flights for a total of 55.4 hours were flown to acquire the data presented within this report.

# TEST AND EVALUATION

## PERFORMANCE

### Hover

Hovering performance was evaluated, using tethered hover techniques, at wheel heights of 150, 100, 80, 47, 22, and 10 feet. Two locations, Bishop Airport (4,100 feet) and nearby Coyote Flats (10,000 feet), were used so that a sufficient temperature range was available to test four different rotor blade tip Mach number ($M_{TIP}$) values (0.60, 0.62, 0.64 and 0.66). The data are presented in nondimensional form, thrust coefficient ($C_T$) versus pressure coefficient ($C_P$) for lines of constant $M_{TIP}$ (figures 1 to 6, appendix), and wheel height versus $C_P$ for lines of constant $C_T$ (figures 7 to 10, appendix). Plots present previous acquired data (reference 3) as well as the results of the test program reported herein. These data should be used to update the HH-53C Flight Manual (reference 2). **(R 1)**[1]

### Takeoffs

Takeoff data were obtained by using a level acceleration takeoff technique at wheel heights of 5 and 15 feet. The 15-foot wheel height was used to simulate the carrying of an external sling load. All flights were made with a mid cg and with the landing gear down. The distance required to clear a 50-foot height was obtained through a range of $\Delta C_P$'s for the two techniques by varying the gross weight of the aircraft from 36,500 to 40,500 pounds. A Fairchild Flight Analyzer was used to record time, height and distance of each takeoff. Figures 11 through 20 of the appendix present these data.

The helicopter was hovered at the two wheel heights using 105-percent rotor rpm, and the power required to hover was noted. The helicopter was then accelerated at the hover height as rapidly as possible to an aim airspeed by pulling the collective to maximum power available or until the rotor rpm drooped to 100 percent. When the helicopter reached the desired indicated airspeed in a level acceleration, the airspeed was maintained in a climb until the helicopter or the simulated sling load cleared the desired 50 foot altitude above the ground. The power required at this point was noted. The difference between the power required in hover and at 50 feet was the $\Delta C_P$ against which takeoff distance was plotted in the previously mentioned figures. For purposes of the test a pace vehicle was used during the acceleration and climb at speeds below 35 knots. Figures 15 and 20 show that the minimum takeoff distance occurs when accelerating to 25 knots and climbing out, while maintaining this speed. This airspeed is not recommended because of inherent inaccuracies in the airspeed indicator at low airspeeds and the large attitude changes required in rotating from a nose low attitude for level acceleration to a nose high attitude for a low speed climb. For an opera-

[1] Boldface numerals preceded by an R correspond to the recommendation numbers tabulated in the Conclusions and Recommendations section of this report.

tional takeoff situation it is necessary to compromise between maximum performance and safety. If a climbout is attempted at 25 knots, a slight over-rotation or a minor attitude change during the climb can cause the airspeed to drop off to such a point that the aircraft can no longer climb and will settle back to the ground. From pilot experience during the takeoff tests, 35 KIAS was the lowest indicated airspeed that was stable enough to be utilized during the takeoff climb.

The acceleration close to the ground and climbout at 35 KIAS provided for a much safer operation. The climb airspeed was very close to the minimum single engine speed of approximately 40 KIAS. A maximum performance takeoff must necessarily be accomplished within the avoid area of the height-velocity diagram; however, single engine speed could possibly be attained if the climbout was at 35 KIAS or a much more controlled touchdown could be made if necessary. For these reasons it is recommended that level acceleration takeoff data be presented using 35 KIAS for the climb. (R 4)

Height-velocity tests have never been conducted at weights greater than 37,000 pounds or at high altitude. A qualitative evaluation of minimum single engine airspeed was made and found to be approximately the same (40 KIAS) as at lower elevation. Single engine out performance should be defined at high altitudes. (R 5)

### Sawtooth Climbs

Sawtooth climbs were made with 100- and 105-percent rotor speeds at approximately 41,000 pounds gross weight. At a density altitude of approximately 11,300 feet collective limits reduced the rate of climb while operating at 100-percent rotor speed as compared to the rate of climb at 105 percent. Figure 21 presents this data. It is recommended that a rotor speed of 105 percent be used when operating at high altitudes and heavy gross weight combinations. (R 2)

### Approaches

High altitude approaches were not specifically evaluated during the performance tests, but enough general observations were made during normal day-to-day operations at the test site to form qualitative opinions concerning the best technique to use when making approaches into a high altitude site at high gross weights. Numerous approaches were made to the high altitude site at 42,000 pounds gross weight and with power margins as low as 10-percent torque (20-percent torque total). Approaches were made to both running landings and a low hover. Approaches were made at a higher true airspeed (same indicated airspeed) than at lower altitudes and caused increased difficulty in slowing the helicopter during the appproach, particularly to a hover. Approaches to a running landing are recommended when a suitable landing area is available and excess hover power is marginal. Running landings with touchdown speeds in the normal 20 to 40 knot range resulted in noticeably higher ground speeds than experienced at lower elevations. This must be anticipated because of the inadequacy of the HH-53C wheel brakes. If a running landing is made and the brakes must be used to stop the helicopter, very short brake life can be expected, especially if minimum distance stops are made with the helicopter at high gross weight. The helicopter can be effectively slowed to a stop by proper application of collective and cyclic controls.

Approaches to a hover were made using a slow, shallow approach to prevent buildup of excessive descent rates or a requirement for a flare to slow the helicopter during the last portion of the approach. There was a tendency for the helicopter to "fall through" as translational lift was lost. This momentary loss of lift was anticipated by insuring that near hover power was applied as the helicopter passed through translational lift, thus reducing the requirement for a sudden application of collective and possible rotor ($N_r$) droop. A rotor speed of 105 percent was used on all heavy weight, high altitude approaches to allow a margin above 100-percent $N_r$ in case the rotor speed was inadvertently drooped when increasing the collective to maximum power. It is recommended that all high altitude, heavy weight approaches be made using maximum rotor speed (full throttles/105-percent) to allow use of maximum power and to avoid reaching an aircraft mechanical limit (tail rotor or collective). (R 2)

## DIRECTIONAL CONTROL EVALUATION

As previously stated the test helicopter was equipped with a new tail rotor bellcrank which increased the control travel of the tail rotor blades by approximately 10 percent. Originally, the tail rotor blade travel was from -3.5 degrees (full right rudder pedal, full down collective) to +24 degrees (full left rudder pedal, full up collective). The new bellcrank maintained the full right pedal, down collective tail rotor blade angle of -3.5 degrees, but increased the full left pedal, up collective blade angle to +27.25 degrees.

Two different methods were used to evaluate the effectiveness of the new tail rotor bellcrank: (1) Pacing of an in-ground-effect (IGE) flight, from left sideward through rearward to right sideward flight, and (2) tethered hover in winds of various velocities.

### Sideward and Rearward Flight

Sideward and rearward flights were conducted at a mid cg (340 inches) 39,000 pounds gross weight and forward cg (328 inches) at 37,500 pounds gross weight. Rotor speeds were 102 and 100 percent, respectively. In addition, left and right sideward flight at 97-percent rotor speed, mid cg (340 inches) and 39,000 pounds gross weight were conducted to compare rotor speed effects. Figures 22 through 28 present these data.

The discontinuities present in the data fairings represent the speed (approximately 20 knots) at which translational lift becomes evident. Comparing sideward flight performance at 102 and 97 percent rotor speed, it is evident that variations in rotor speed significantly influence the aircraft's capabilities. The conventional bellcrank blade angle limits are included in the plots. It can be seen that the new bellcrank significantly increased the sideward flight capability. Where two data points are presented they represent the absolute maximum and minimum values recorded during that particular data point. At 97-percent rotor speed the aircraft is collective limited in left sideward flight and tail rotor blade limited in right sideward flight. The lower rotor speeds increased cruise guide readings, as was expected (reference 2). Higher rotor speeds (above 100-percent, see section Tethered Hover) resulted in more normal response higher directional control capability, and reduced cruise guide readings. When operating the HH-53C at high altitudes, 105-percent rotor speeds should be used. (R 2)

### Tethered Hover

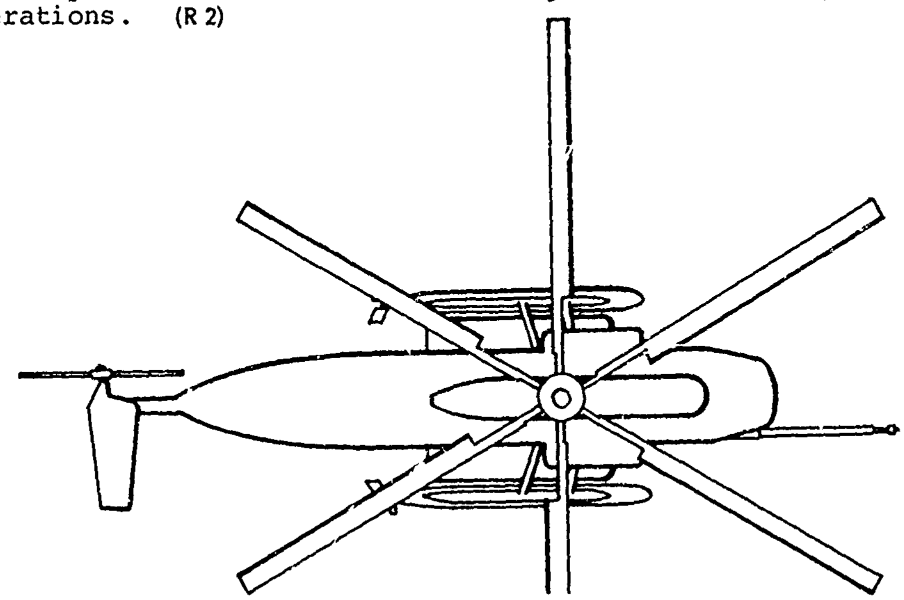
Hover data were obtained using tethered hover techniques. Increasing gross weights up to the limit gross weight were simulated by pulling on a cable tethered to the ground. Power was increased until reaching the gross weight limit or an aircraft power or control limit.

Hover testing was performed at Coyote Flats and Bishop Airport.

While hovering in a wind, various aircraft headings relative to the wind could easily be established. Then, by monitoring tail rotor torque and blade angle, a critical relative heading was determined. At approximately 200 degrees relative heading the HH-53C experienced a shuddering, similar to that felt when translational lift is lost during approach to a hover. However, maximum blade angle and tail rotor torque were obtained at approximately 80 degrees relative heading.

Tethered hover in little or no wind (3 knots or less) showed the new tail rotor bellcrank to be sufficient for $C_T$ values up to 107 (figures 30 through 33). At 9,500 feet pressure altitude, the new bellcrank increased the HH-53C's capability by approximately 4,500 pounds. The new bellcrank will prevent any tail rotor control problems for the present aircraft limits. It should be utilized to prevent lack of tail rotor control from limiting aircraft performance. (R 3)

During all tethered hover tests, the helicopter's handling and control response were satisfactory. To obtain referred performance data, rotor speed ranged from 95 to 105 percent. No control response problem was encountered even at the relatively low rotor speed of 94 percent. However, at rotor speeds below 100 percent, control response was sluggish and consequently more fatiguing to the pilot when trying to maintain a stable hover. When operating at rotor speeds below 100 percent, collective and tail rotor mechanical limits were occasionally reached. To maintain a more normal control response and insure full utilization of available engine power without being limited by a mechanical control stop a rotor speed of 105 percent should be used for high altitude, heavy gross weight hover operations. (R 2)



# CONCLUSIONS AND RECOMMENDATIONS

The HH-53C helicopter's hover and takeoff performance were satisfactory during high altitude testing. Operating at 105-percent rotor speed at high altitude, heavy gross weight combinations provided a more rapid response of the aircraft to control inputs than operation at 100 percent where control response was somewhat sluggish. In addition, it provided increased performance (more power available), avoided reaching mechanical control stops (collective or tail rotor) which would limit performance, and lowered cruise guide readings. The new tail rotor bellcrank increased the HH-53C lift capability approximately 4,500 pounds by preventing lack of tail rotor authority from limiting aircraft performance.

1. The hover data presented in this report should be used to update the HH-53C Flight Manual (page 2).

2. A rotor speed of 105 percent should be used for high altitude, heavy gross weight combinations (pages 3, 4, and 5).

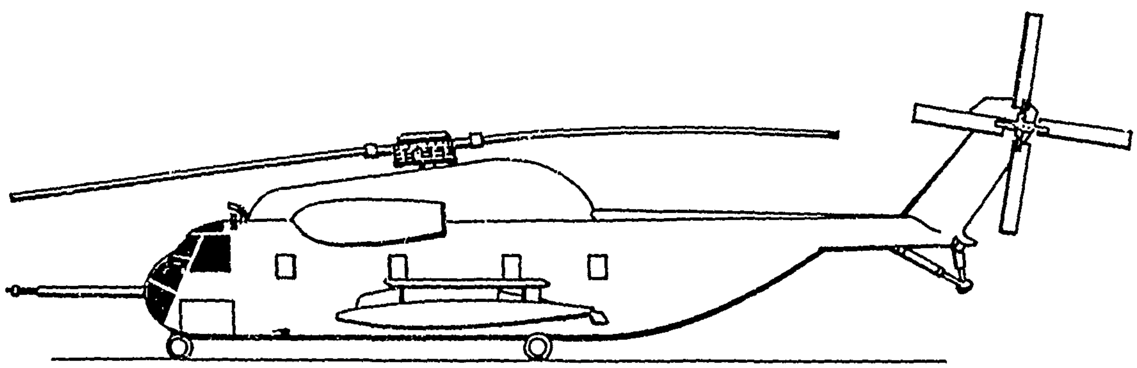
3. The new tail rotor bellcrank should be incorporated to prevent the lack of tail rotor control from limiting the aircraft's performance (page 5).

Maximum takeoff performance over a 50-foot obstacle was obtained by climbing at approximately 25 knots. However, a compromise between performance and safety dictated the use of 35 knots.

4. Takeoff charts should be based on 35-knot climbout speeds (page 3).

Height velocity tests have never been conducted at weights greater than 37,000 pounds gross weight or at high altitude.

5. Single engine out performance should be defined at high altitudes (page 3).



# APPENDIX I

# DATA ANALYSIS METHODS AND TEST DATA

## GENERAL

Dimensional analysis of the major items affecting helicopter performance yields several sets of dimensionless variables which can be used to present performance data in a useful nondimensional form. These variables are defined as follows:

$$C_P = \frac{SHP \times 550}{\rho A\ (\Omega R)^3}$$

$$C_T = \frac{W}{\rho A\ (\Omega R)^2}$$

$$M_{TIP} = \frac{\Omega R}{38.967\ \sqrt{T_a}}$$

Where SHP is in foot-pound x 550 per second, $\rho$ is the air density in slugs per foot cubed, A is the rotor disc area in feet squared, $\Omega$ is the rotor angular velocity in radians per second, R is the rotor radius in feet, and $T_a$ is the temperature of the ambient air in degrees Kelvin.

## POWER DETERMINATION

The HH-53C employed an electronic torque monitoring system to measure the percent of torque being applied by each engine to the main transmission. A torque reading of 100 percent was equivalent to 3,200 SHP. The torque sensing system was located at the engine input section to the nose gearbox and was made up of the torque shaft, torque pickup, the phase detector, and the torque indicator. The torque sensor shaft consisted of an inner and outer shaft arranged so that the inner shaft was subjected to the power turbine load. The major diameter on each shaft was machined to contain 72 teeth. This portion of the shaft was the exciter. The torque pickup was a coil installed in the torque tube opposite the exciter.

The system measured torque by measing the twist in the shaft connecting the engine to the load. To measure this twist a pickup was installed in the torque tube opposite a pair of gear teeth on the rotating shaft. As the shaft rotated, two ac signals were induced in the pickup coils. As the torque in the shaft increased, the two sets of teeth were displaced from each other as the shaft twisted and the phase angle difference between the two ac voltages changed. The output of the pickup coils was fed into a phase detector that electronically measured the phase angle change. This phase angle was then converted to an output

voltage proportional to torque. Test SHP was determined from inflight torquemeter readings and rotor rpm using the following equation.

$$\frac{SHP}{engine} = \frac{(13,600)\ (\%\ rpm)\ (\%\ Q)\ (1,235)}{5,250}$$

Where Q is percent torque reading from the aircraft's instruments.

## SAWTOOTH CLIMBS

The data presented in figure 21 of this appendix are the test day observed rates of climb corrected to tape line rates of climb using the following equation:

$$R/C_t = \frac{dh}{dt} \times \frac{Ta_t}{Ta_s}$$

where $R/C_t$ was the tapeline rate of climb in feet per minute, dh/dt being the slope of the pressure altitude versus time curve in feet per minute. $Ta_t/Ta_s$ was the ratio of the test day ambient temperatures to the standard day temperature for the test altitude.

## CONTROL POSITIONS

The following controls and their positions were monitored during the tail rotor evaluation portion of these tests: lateral and longitudinal cyclic stick, collective stick, and rudder pedals. Sensors were located in the "broom closet" located on the aft side of the bulkhead dividing the cockpit and cargo area. These pickups only registered control inputs. The actual control output was a function of the pilot input, collective-rudder pedal coupling, and the automatic flight control system (AFCS). Consequently, the following additional parameters were also monitored: the AFCS servo output, tail rotor blade pitch angle and the tail rotor torque.

The AFCS's input occurs in series with pilot's input and does not necessarily show as a control position change. It is possible for the pilot to have 20-percent left rudder pedal travel remaining, yet have the tail rotor blade angle at the limit (+27.25 degrees). This phenomenon occurs in the data presented in the tail rotor evaluation. It is a result of the collective-rudder coupling and AFCS authority. Figure 34 shows how collective position can affect rudder pedal position and tail rotor blade angle.

Arbitrarily, 100-percent right rudder pedal was defined as that position where collective was full down and tail rotor blade angle was -3.5 degrees with AFCS off. Left rudder (0 percent) was defined at full up collective with tail rotor blade angle at +27.25 degrees.

Figure 35 shows this situation: full right rudder is applied (full pedal travel; collective control is full down; now, when the collective control is raised, the rudder pedal travel available increases, although the tail rotor blade angle actually <u>decreases</u>. Figure 36 shows the converse of this (up collective, left pedal).

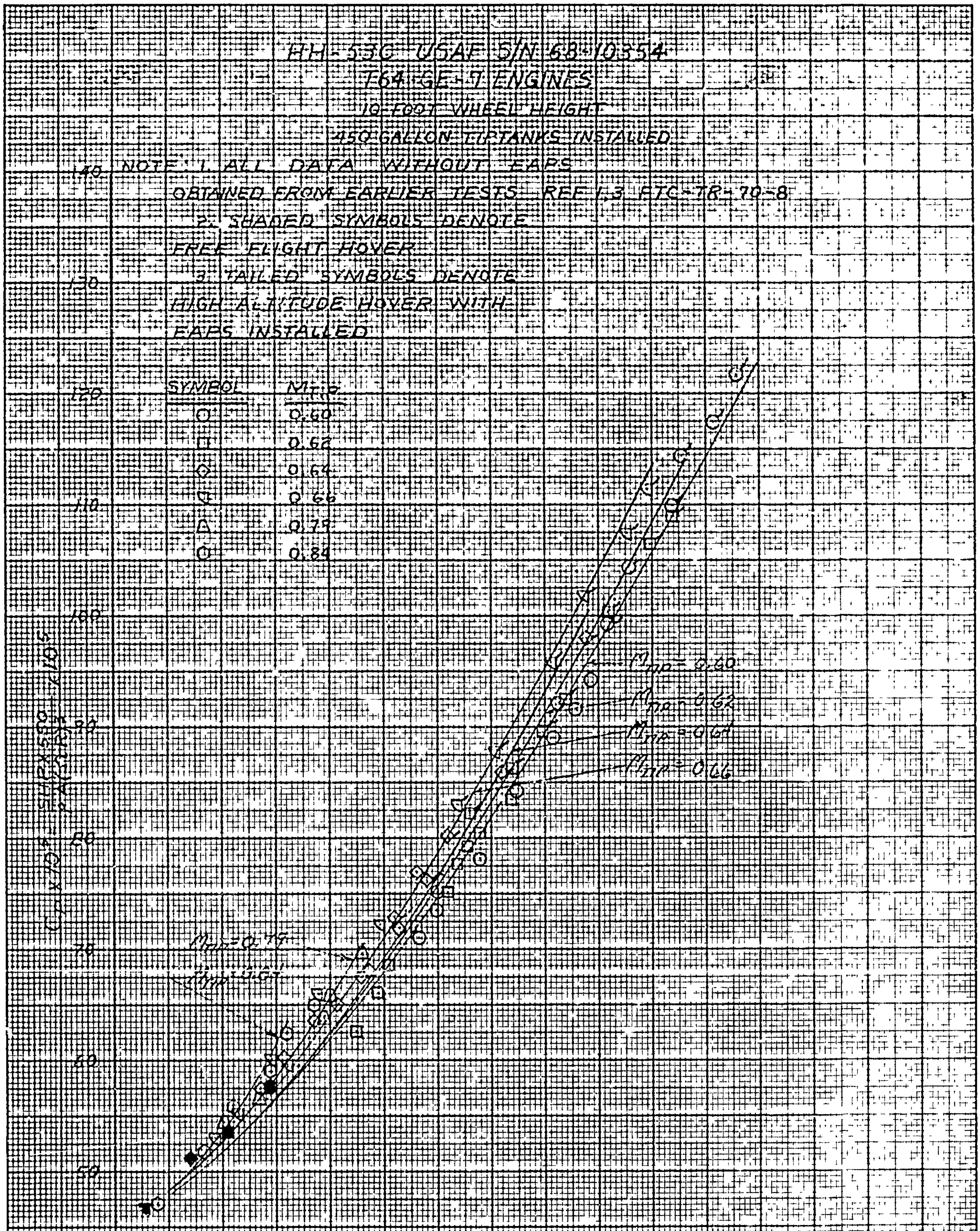



Figure 1 Nondimensional Hovering Performance

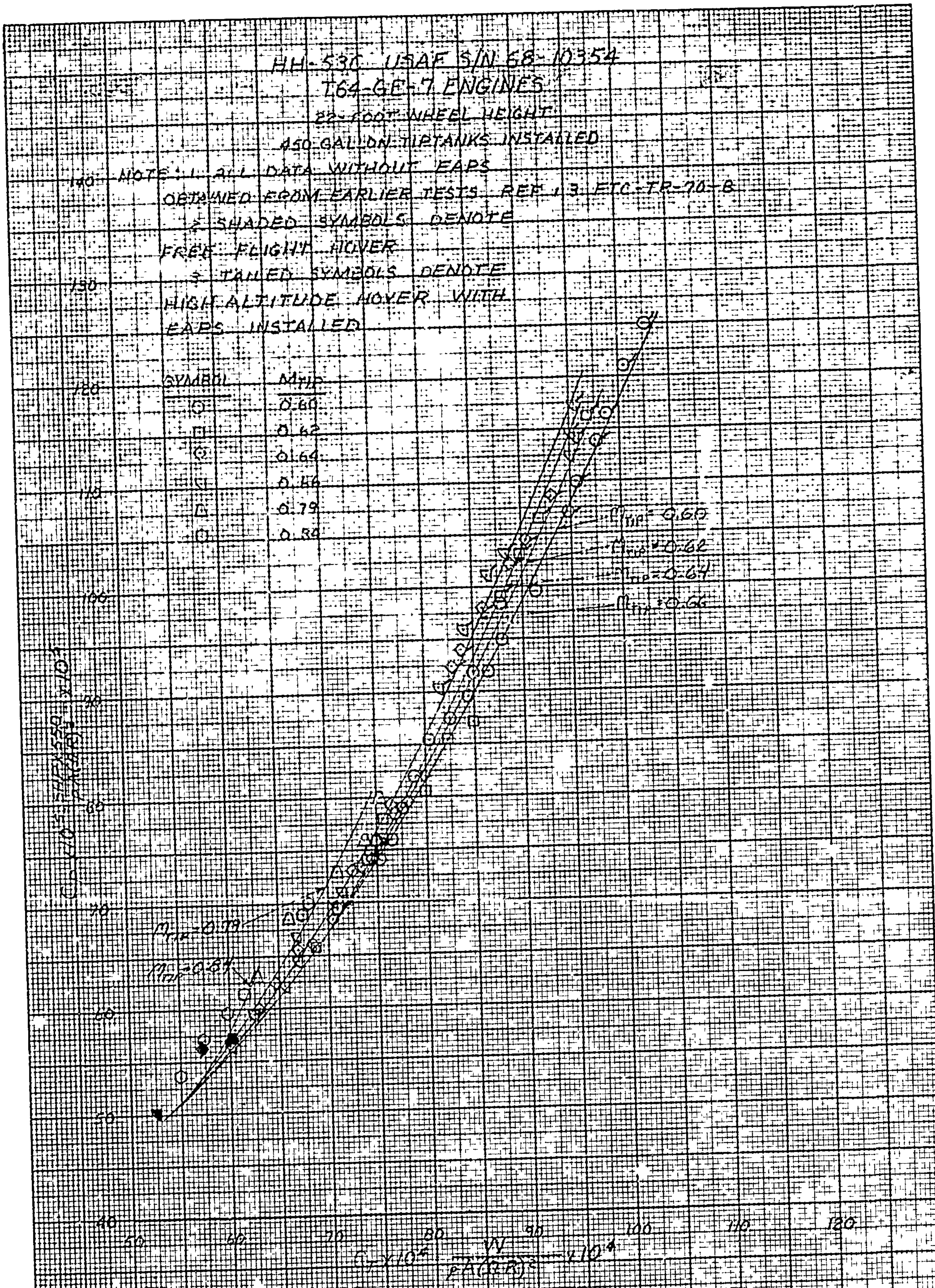



Figure 2 Nondimensional Hovering Performance

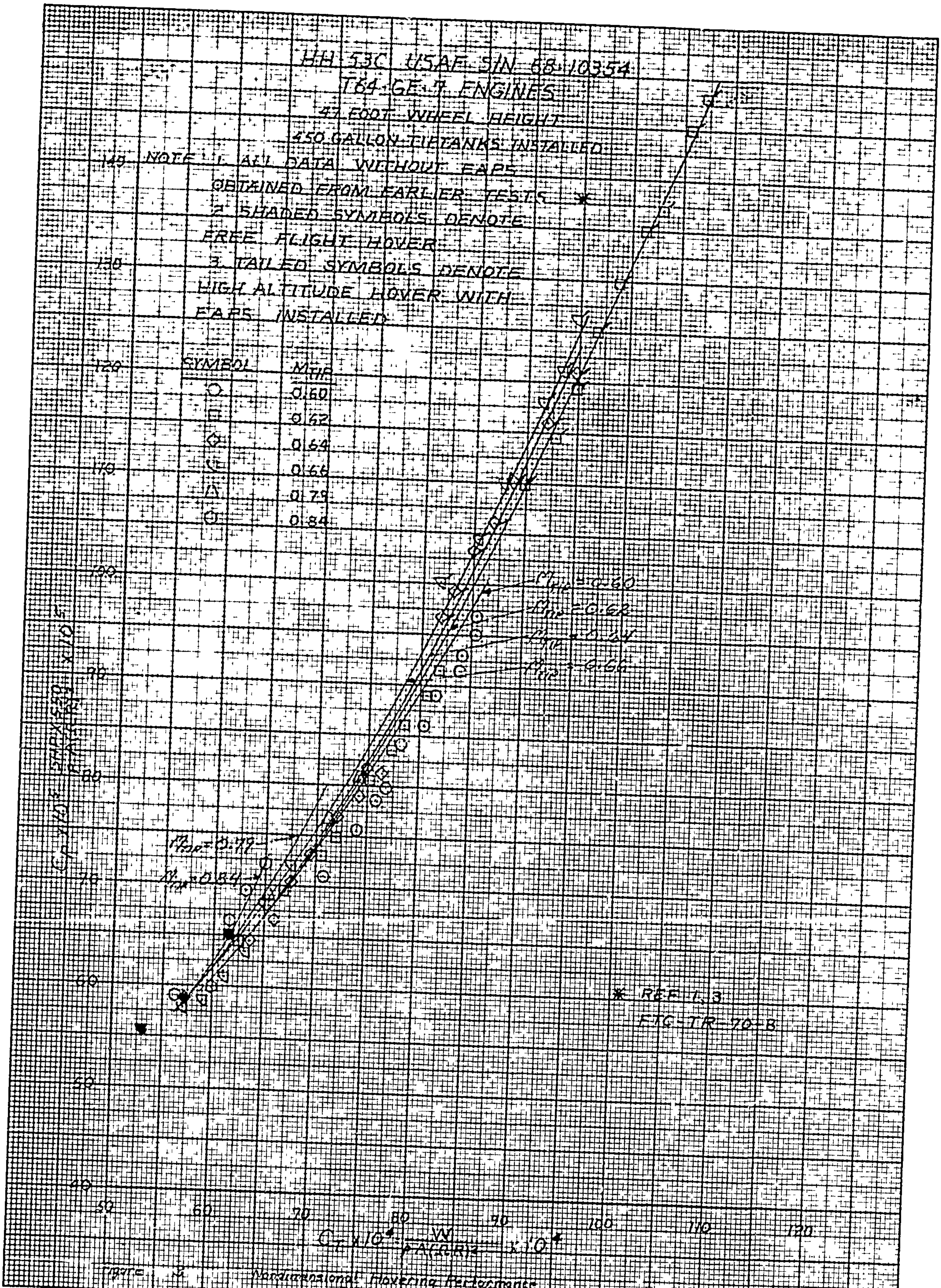



Figure 2 Nondimensional Hovering Performance

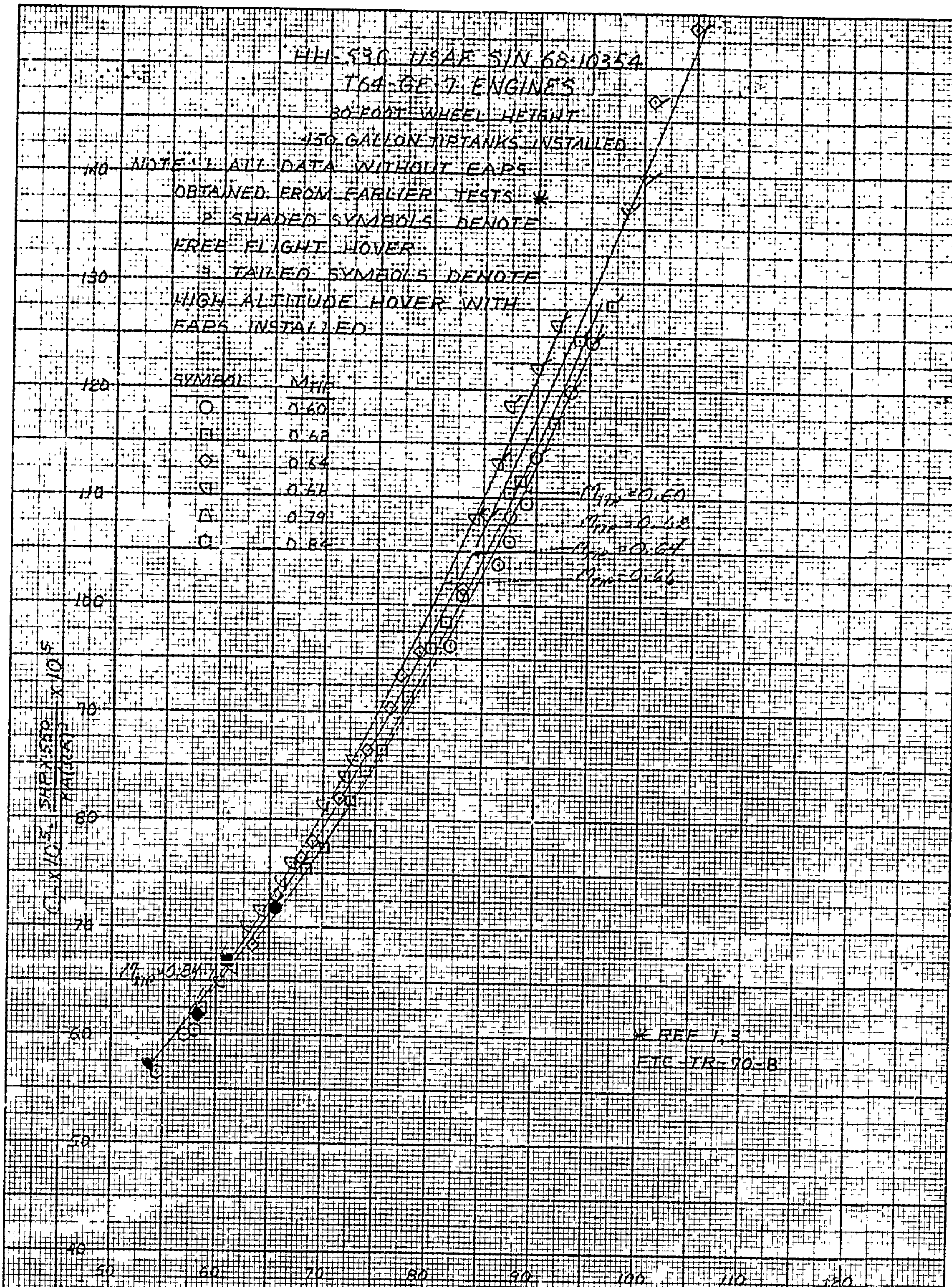



Figure 4 Nondimensional Hovering Performance

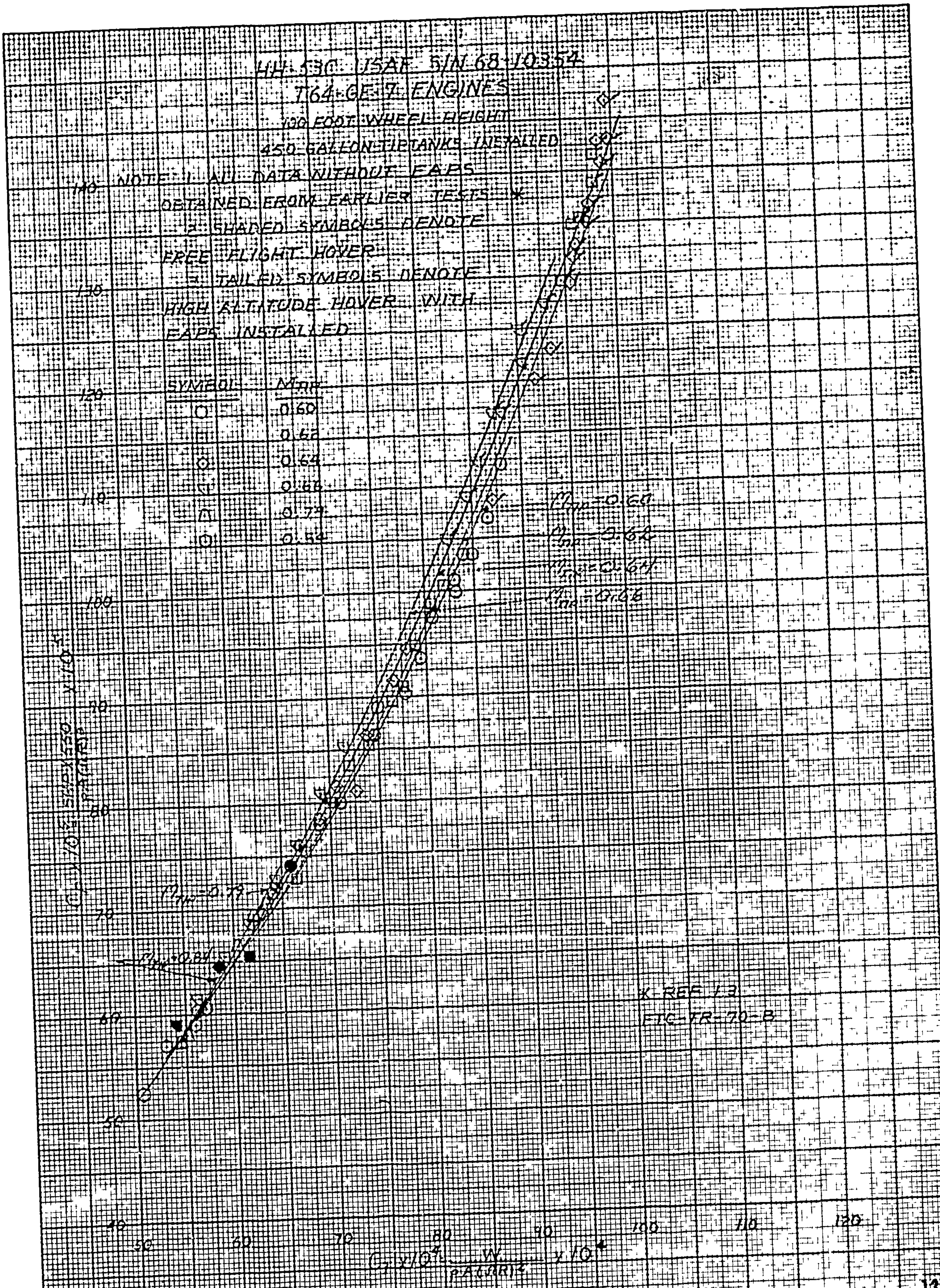



Figure 5 Nondimensional Hovering Performance

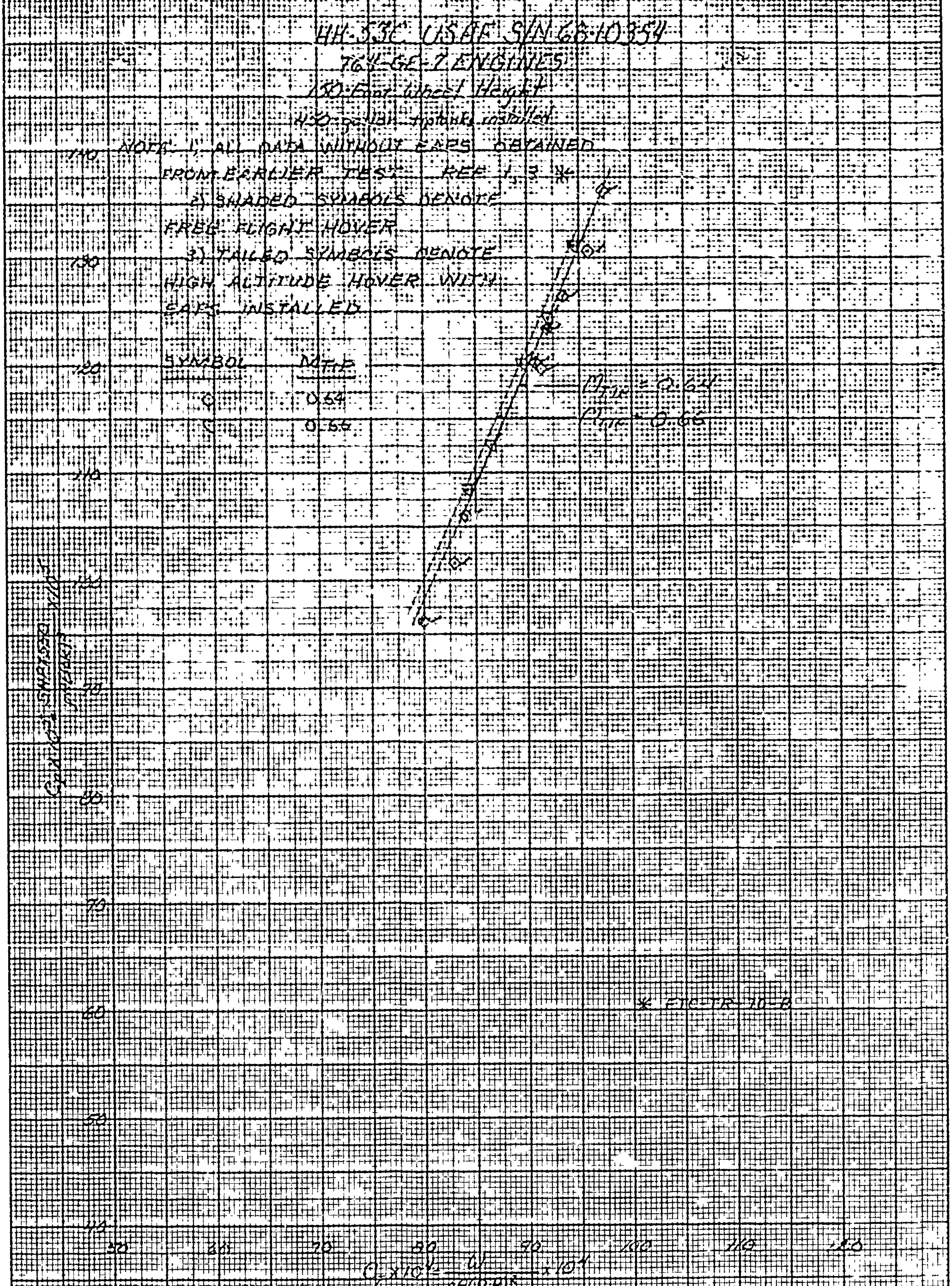



Figure 6 Nondimensional Hovering Performance

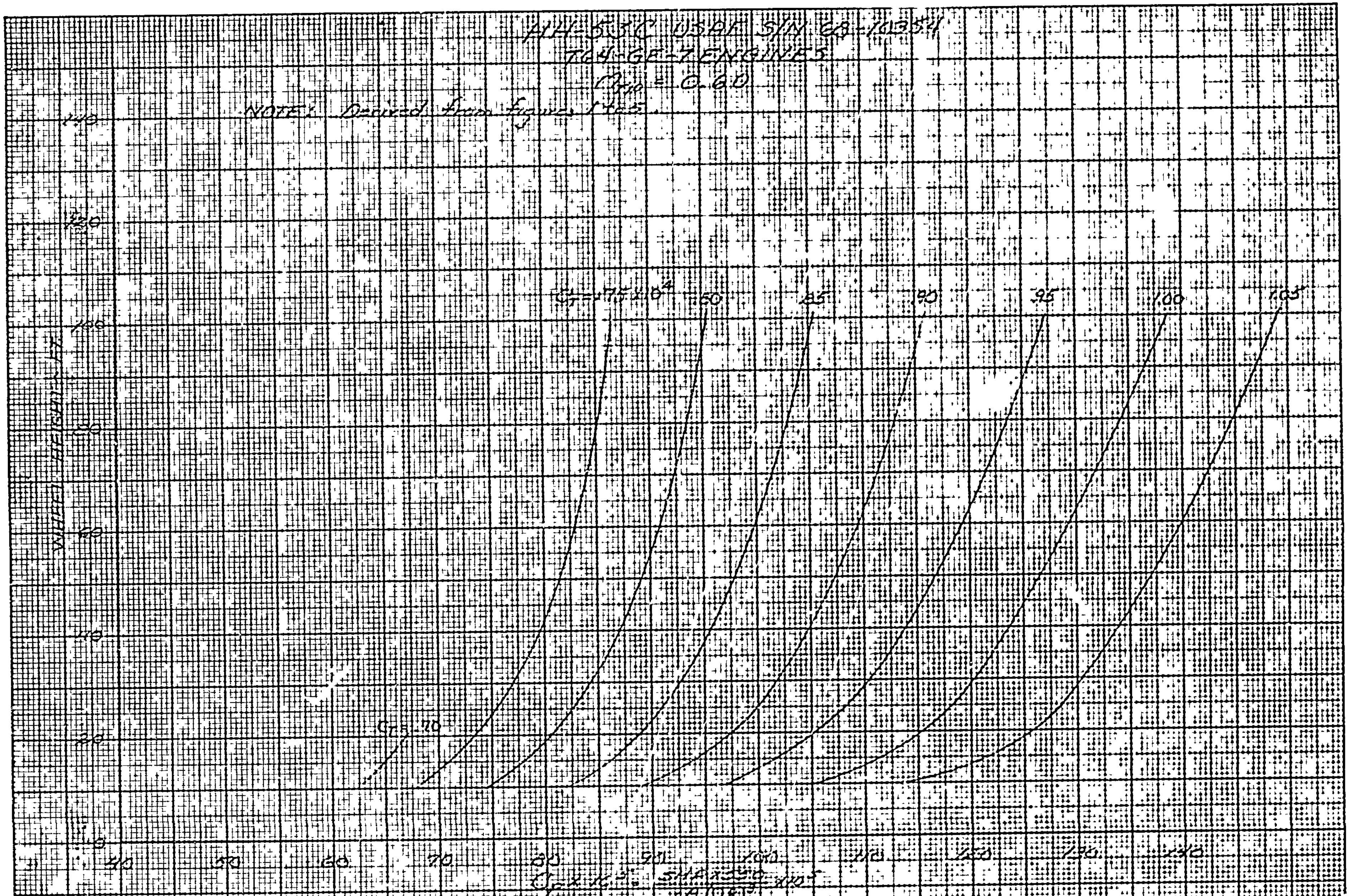



Figure 7 HOVER PERFORMANCE SUMMARY

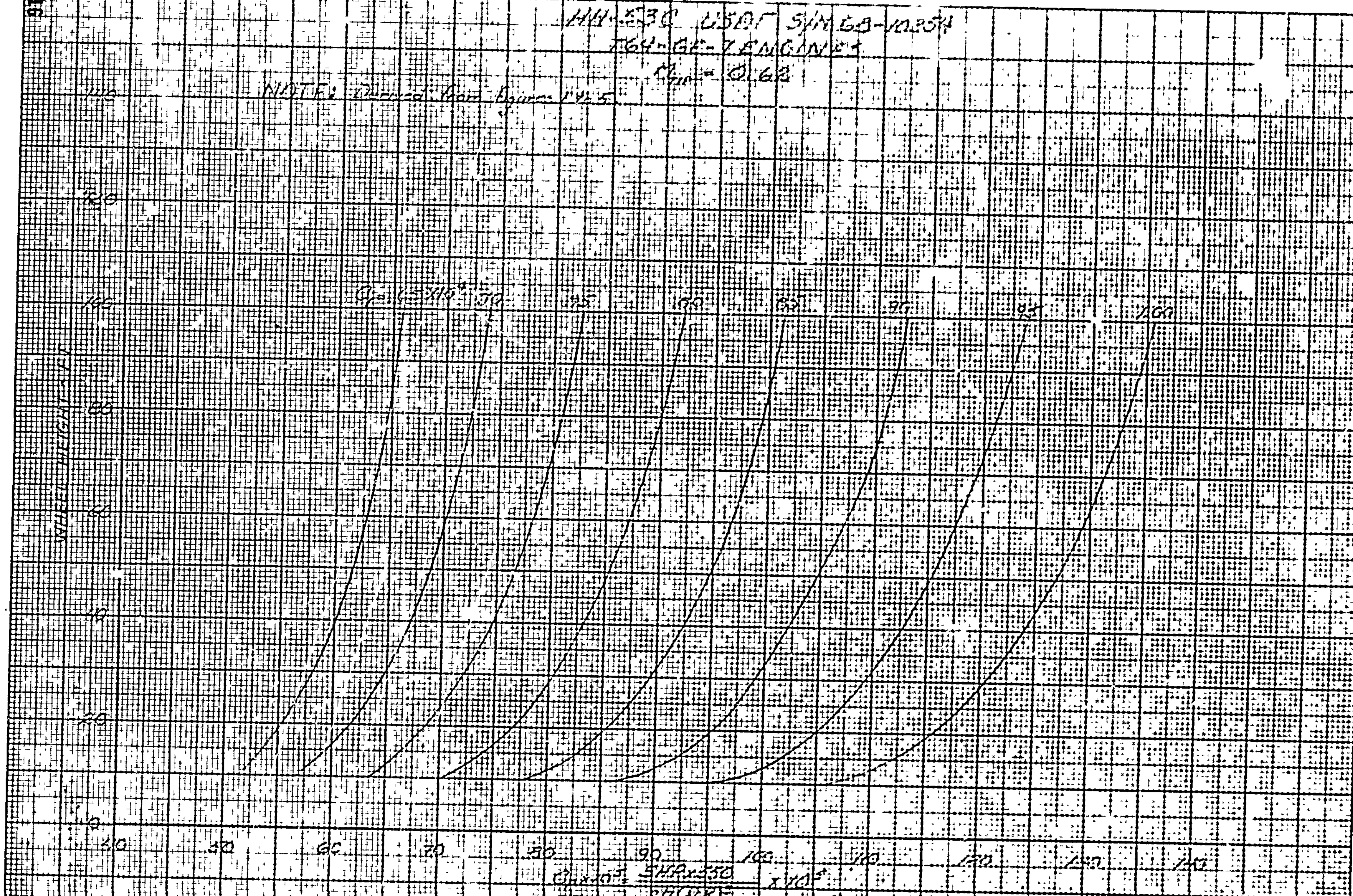



Figure 8 HOVER PERFORMANCE SUMMARY

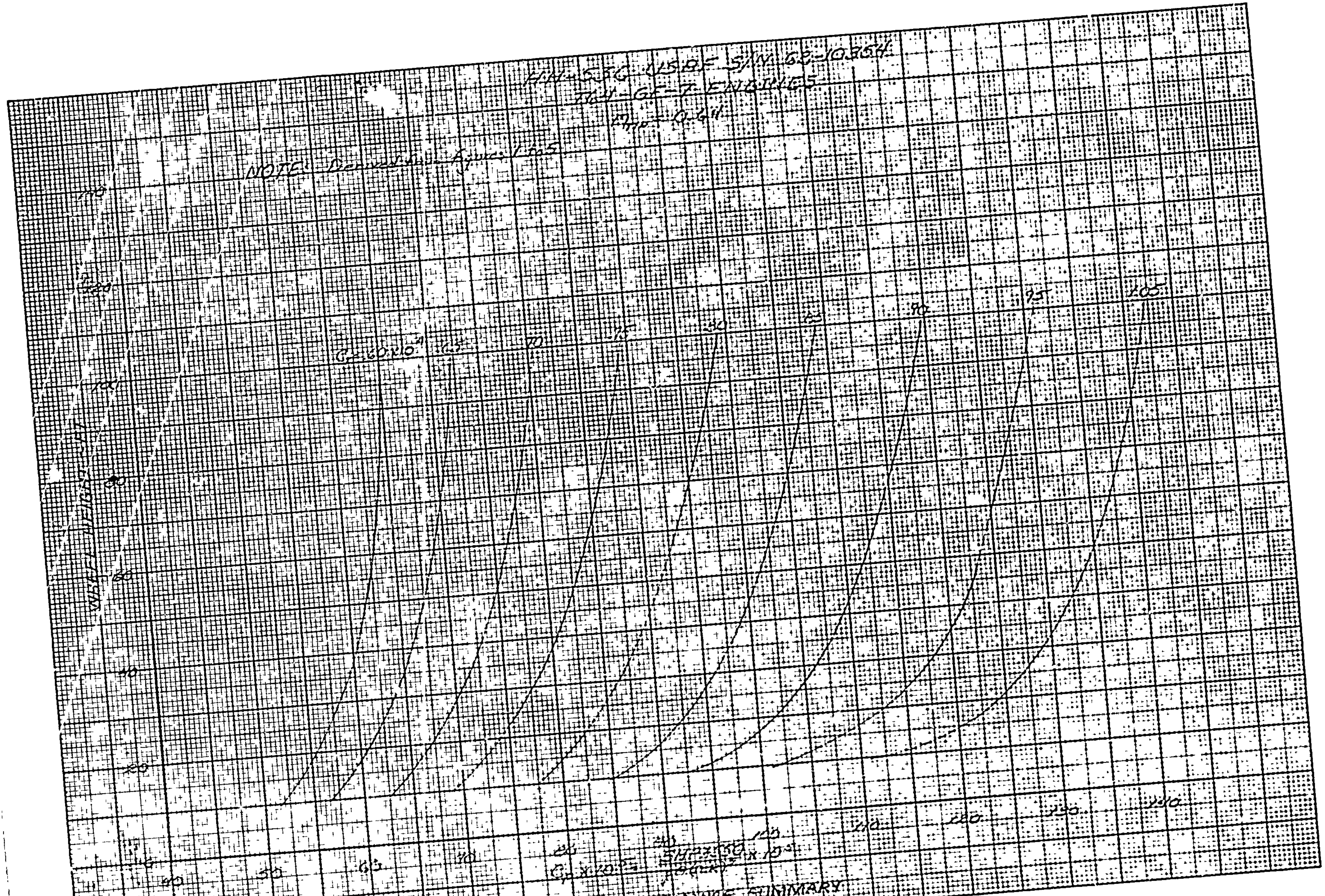

HH-53C USAF S/N 62-10354
T64-GE-7 ENGINES
NOTE
WEIGHT
HOVER PERFORMANCE SUMMARY

Figure 9

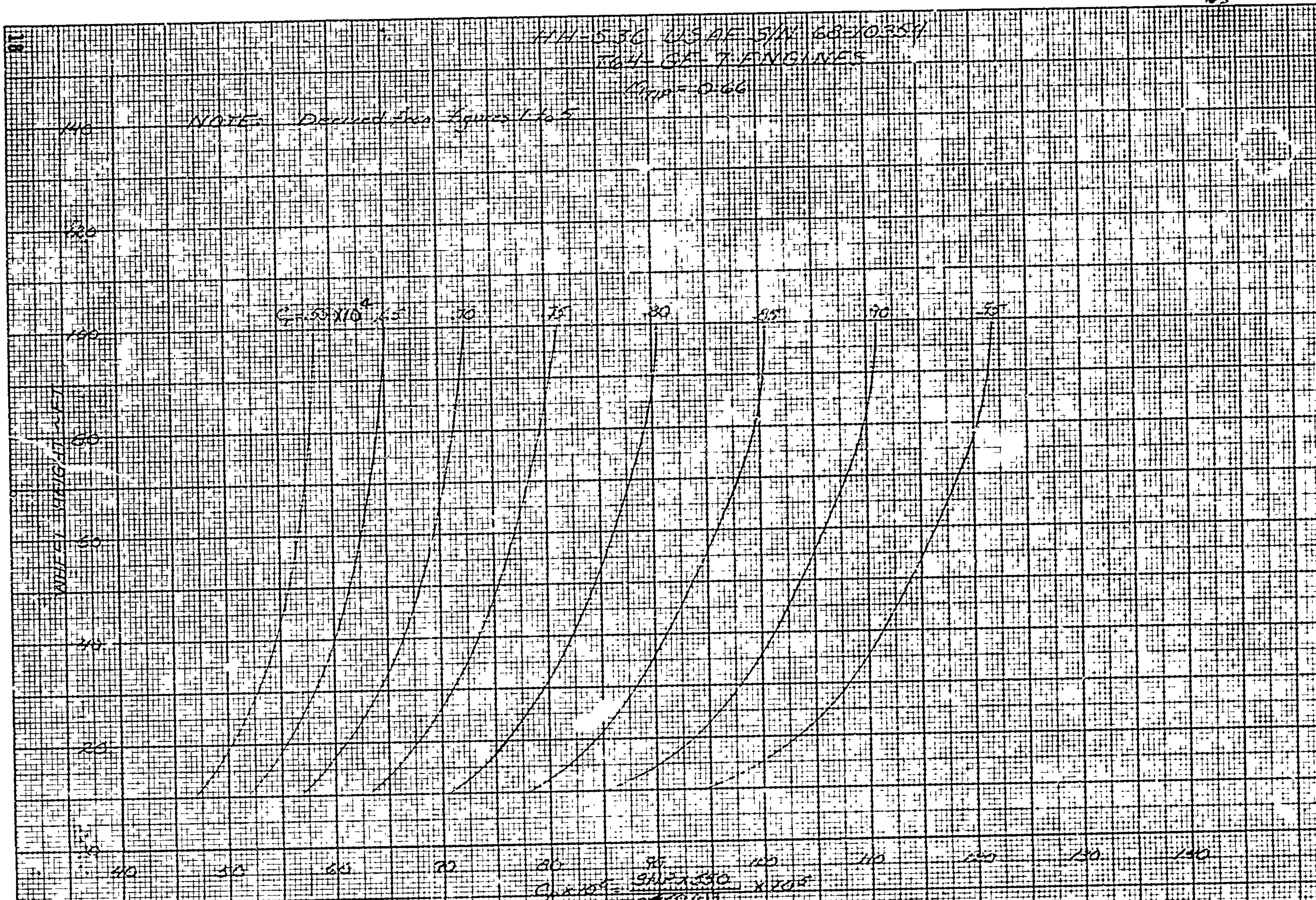
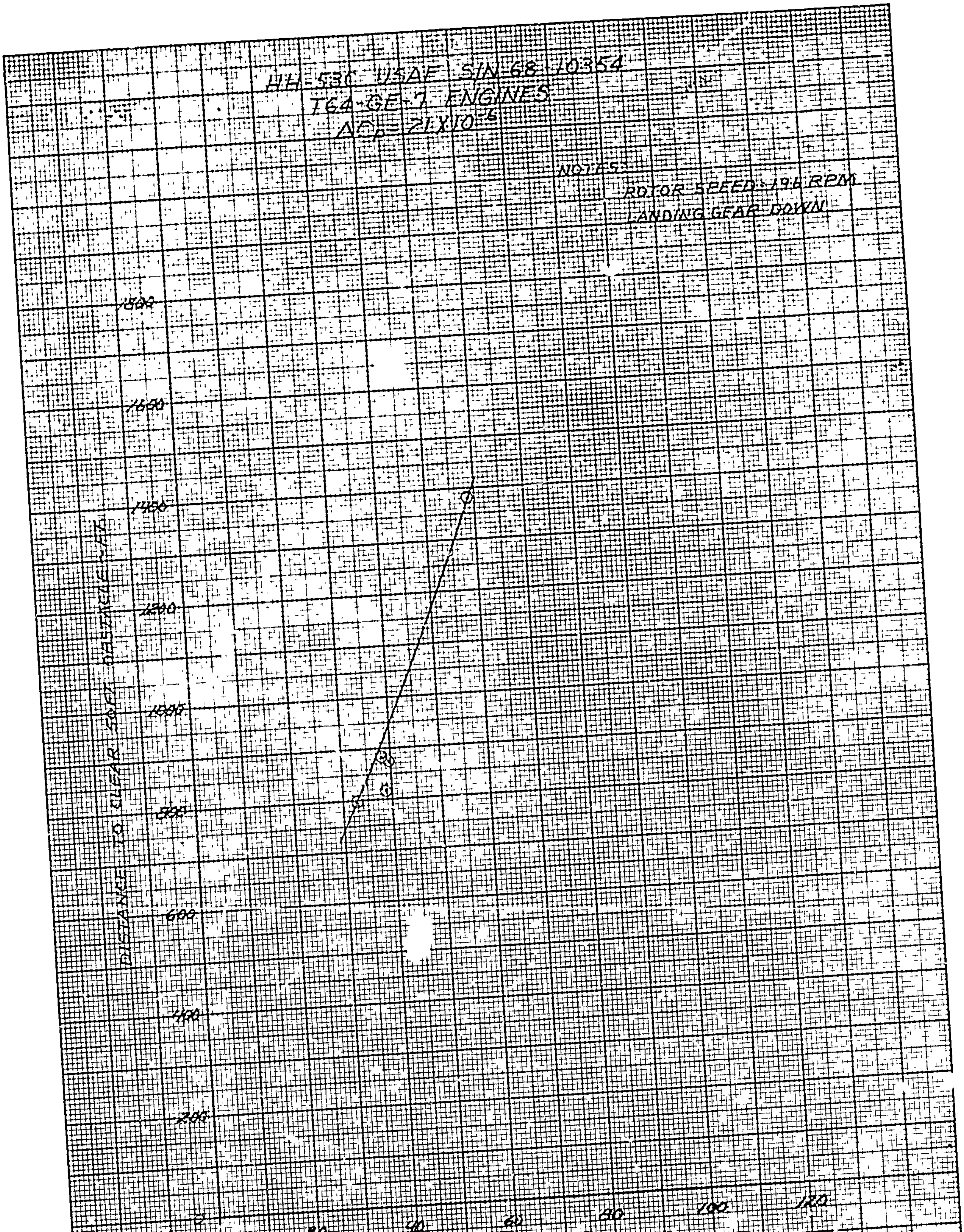
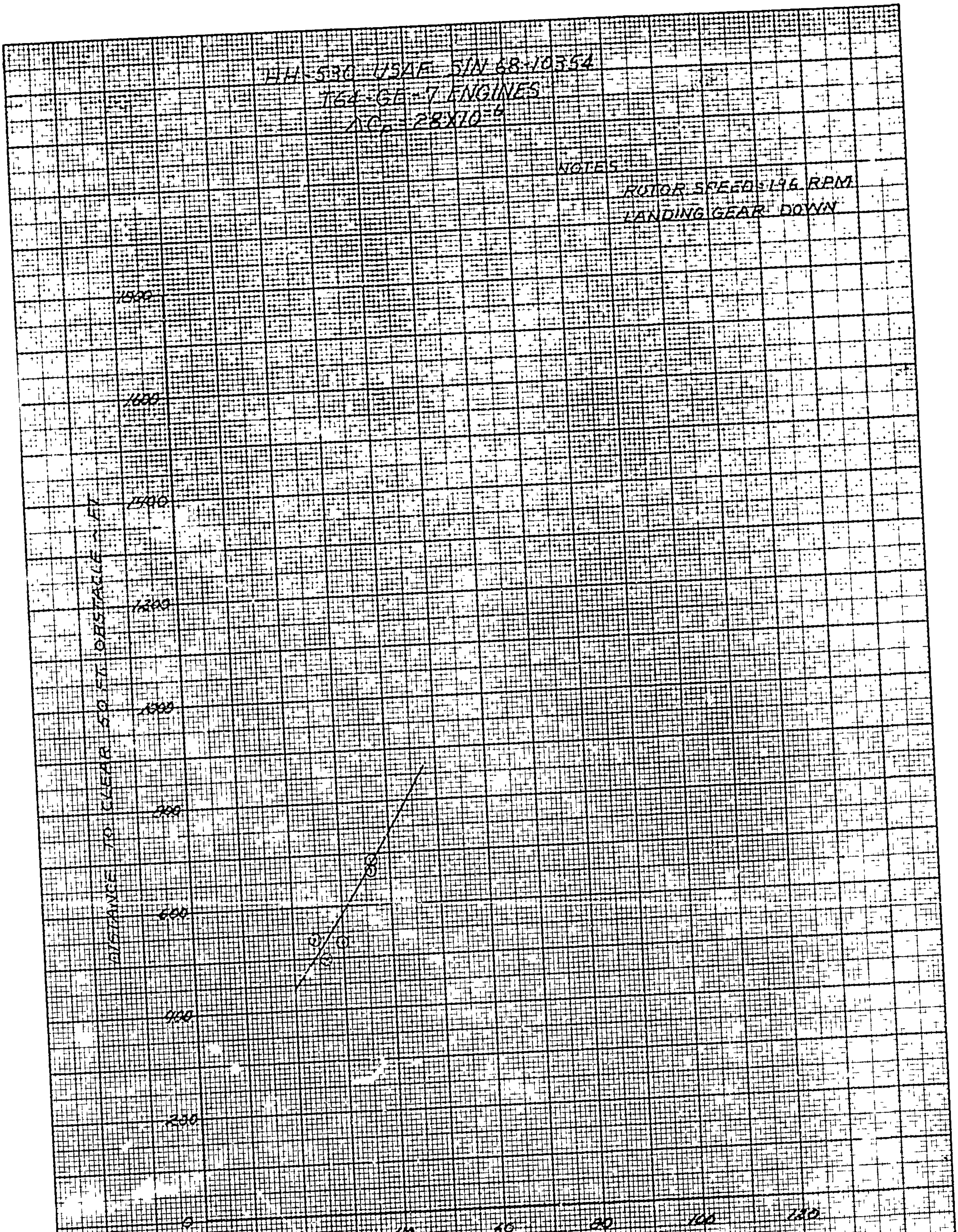


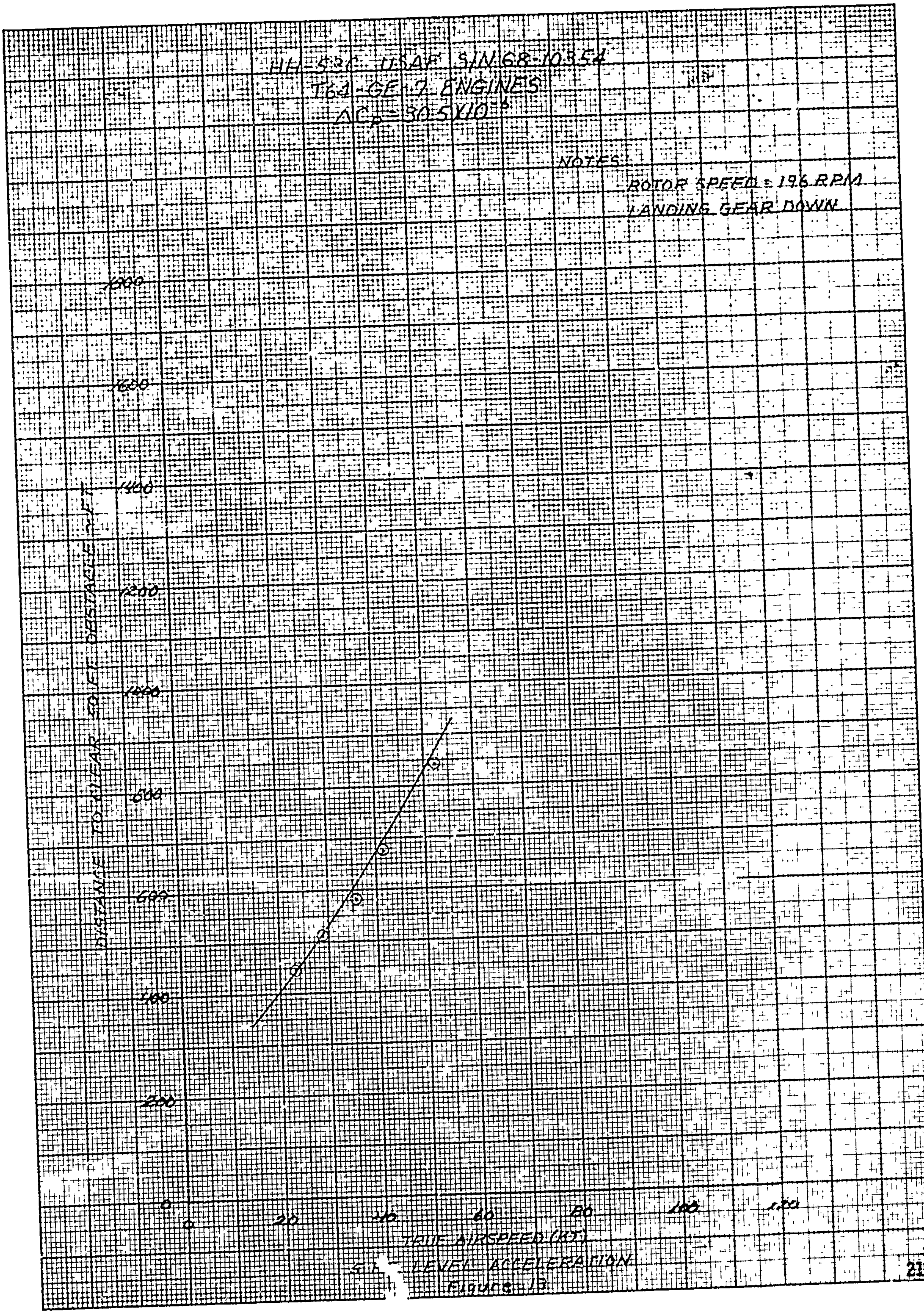

Figure [illegible] HOVER PERFORMANCE SUMMARY



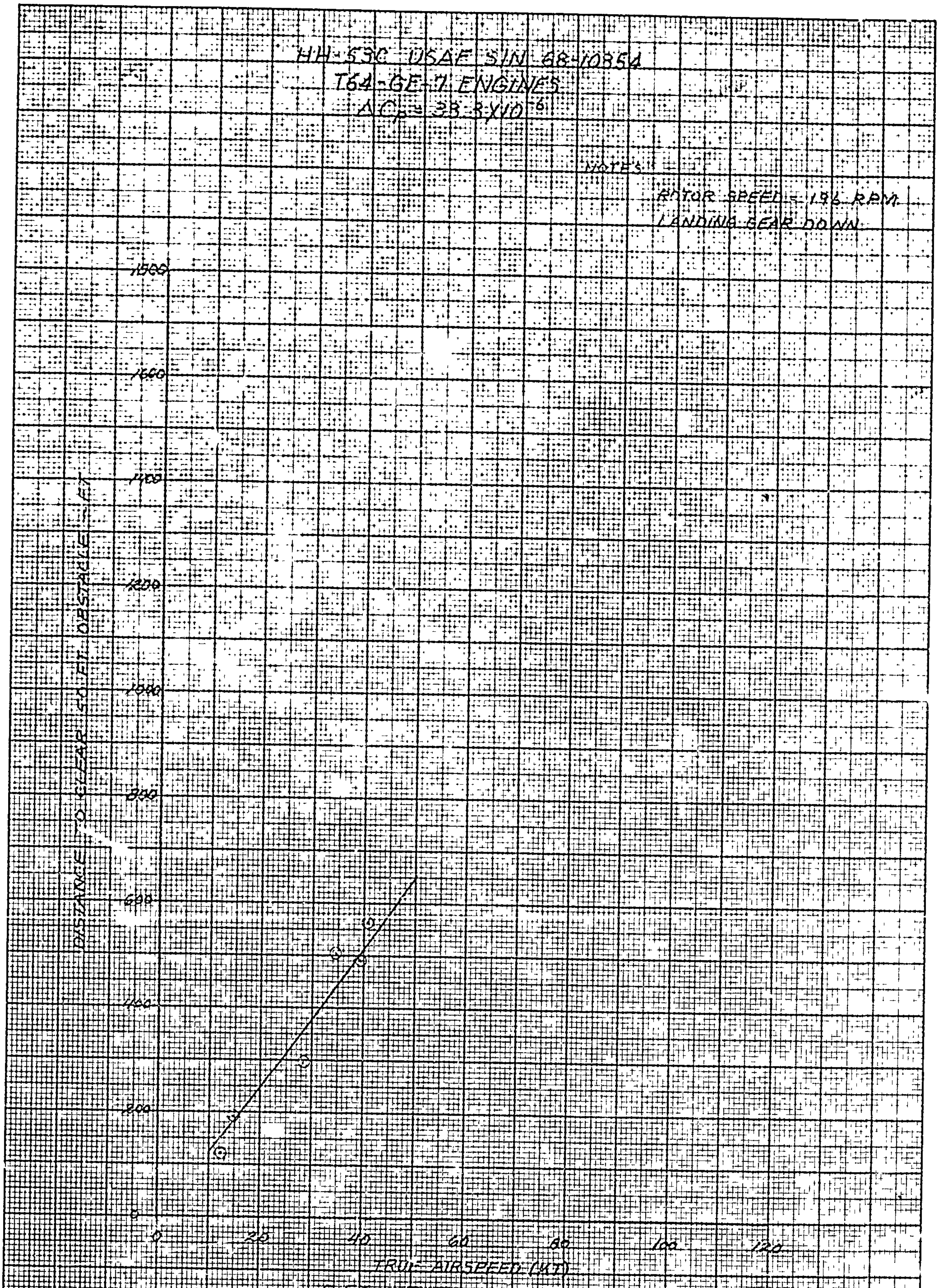

5 FT. LEVEL ACCELERATION
Figure 11



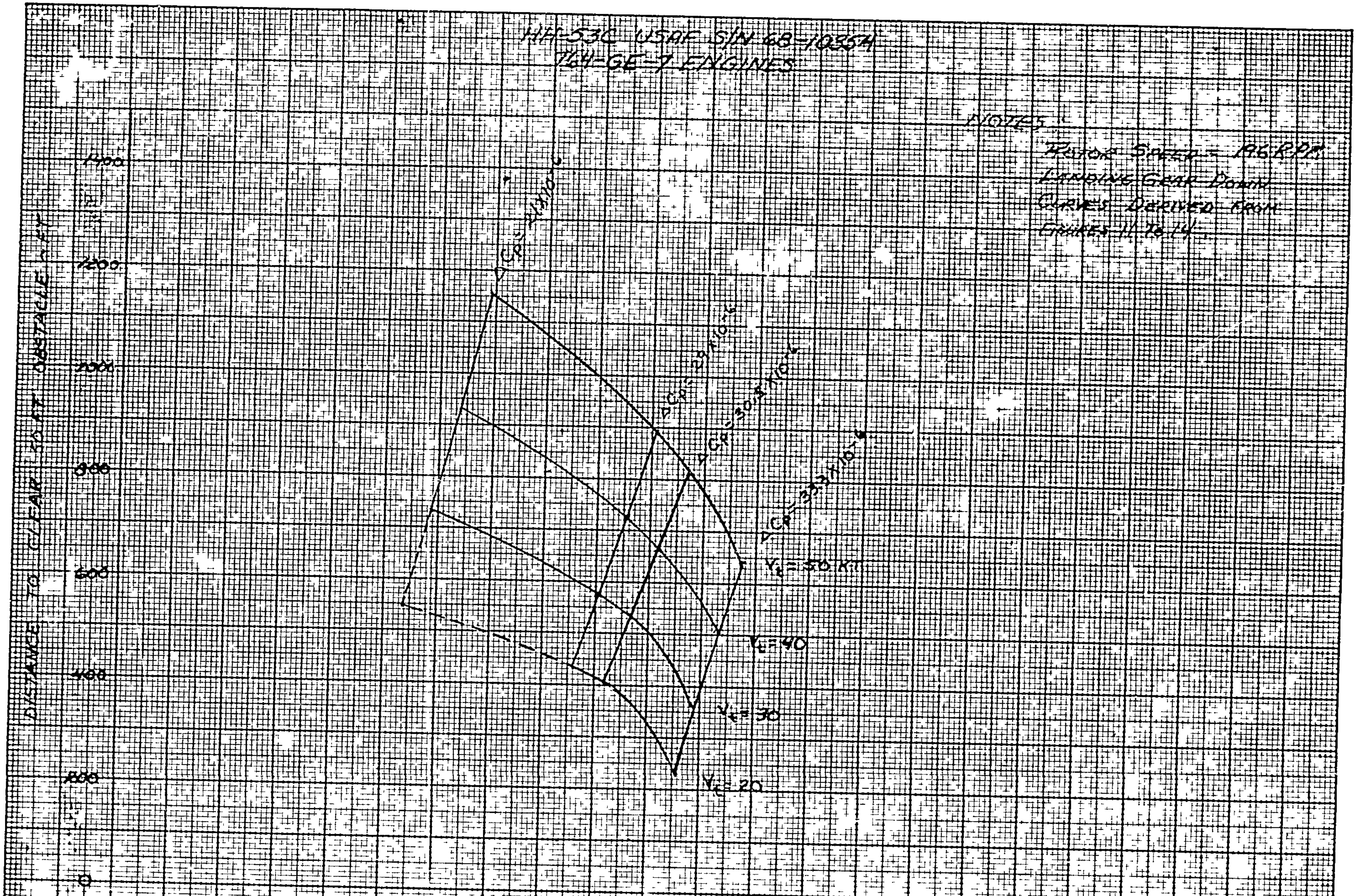

6 FT LEVEL ACCELERATION
FIGURE 12




S.L. LEVEL ACCELERATION
Figure 13

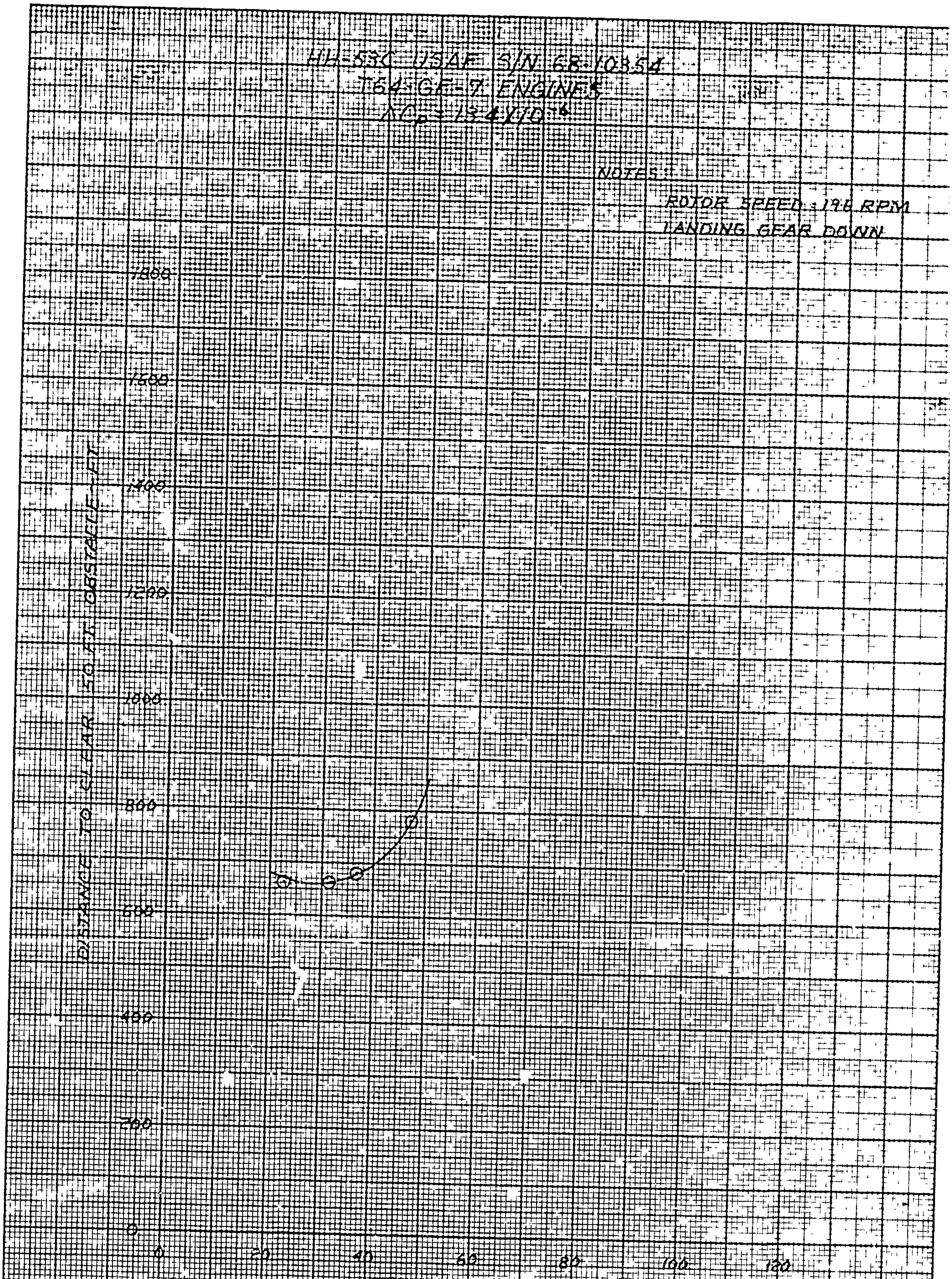



5 FT LEVEL ACCELERATION
Figure 14

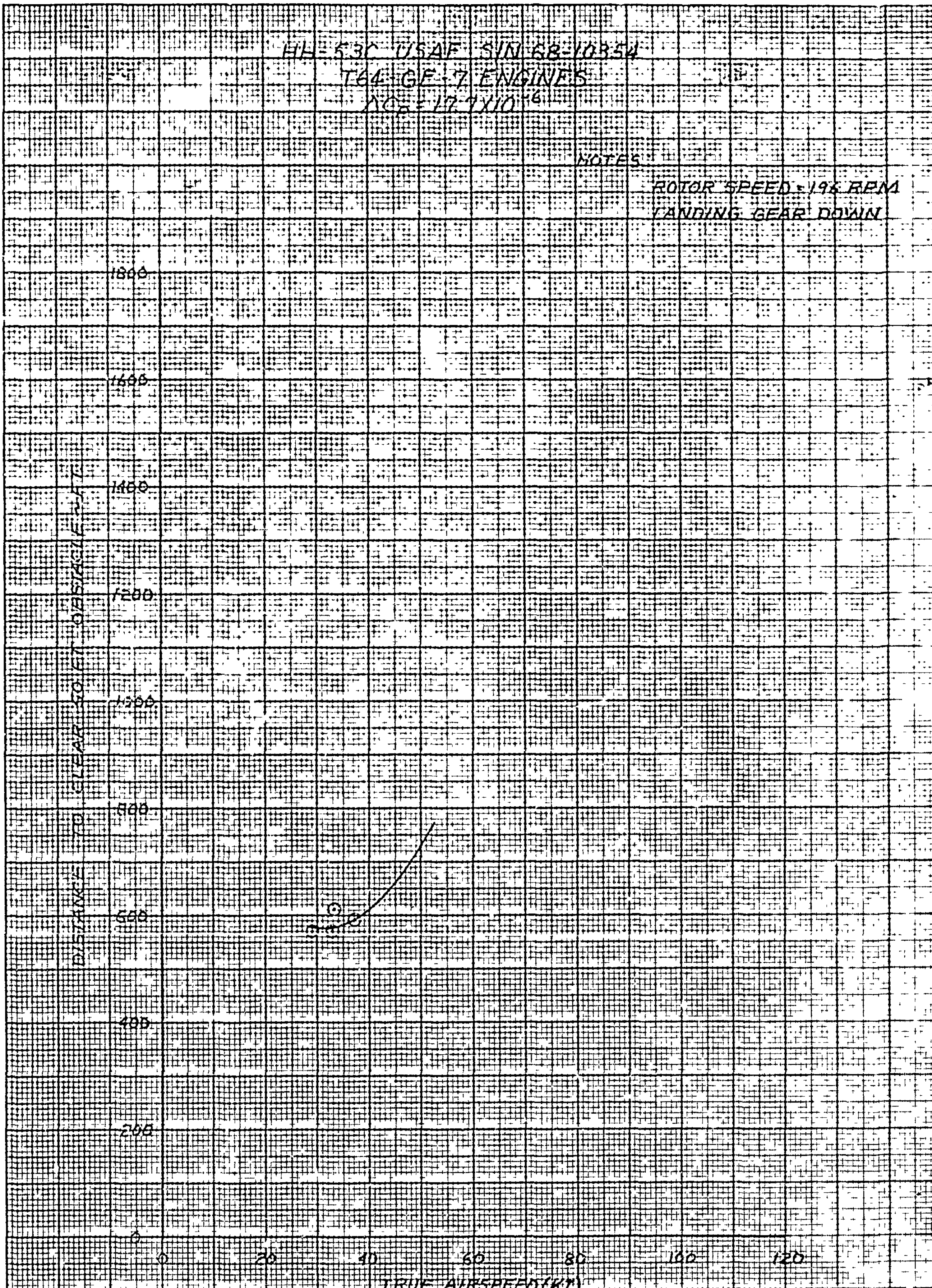



5 Foot Level Acceleration Summary
Figure 15

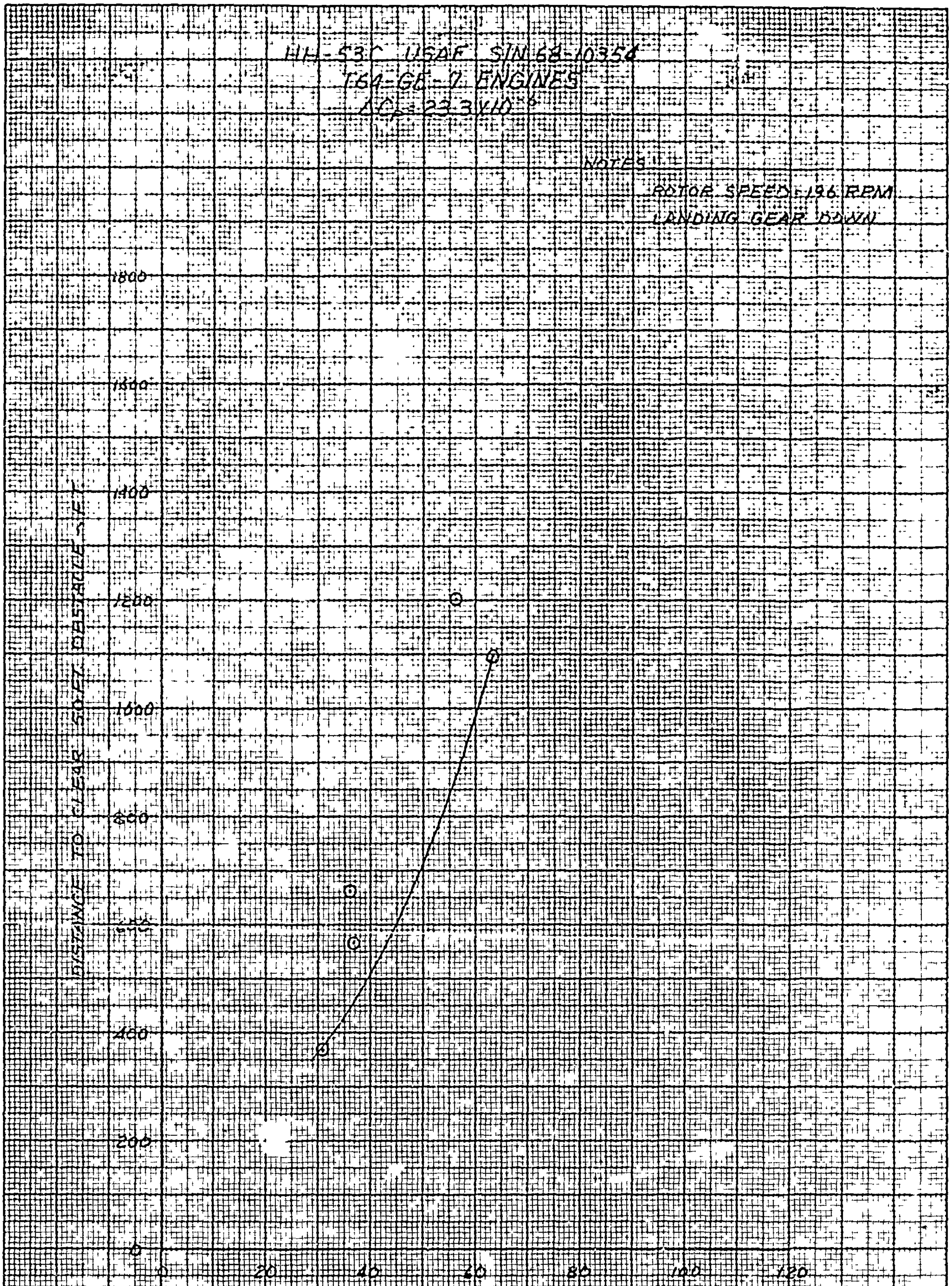

15 FT LEVEL ACCELERATION
Figure 16




15 FT LEVEL ACCELERATION
Figure 17



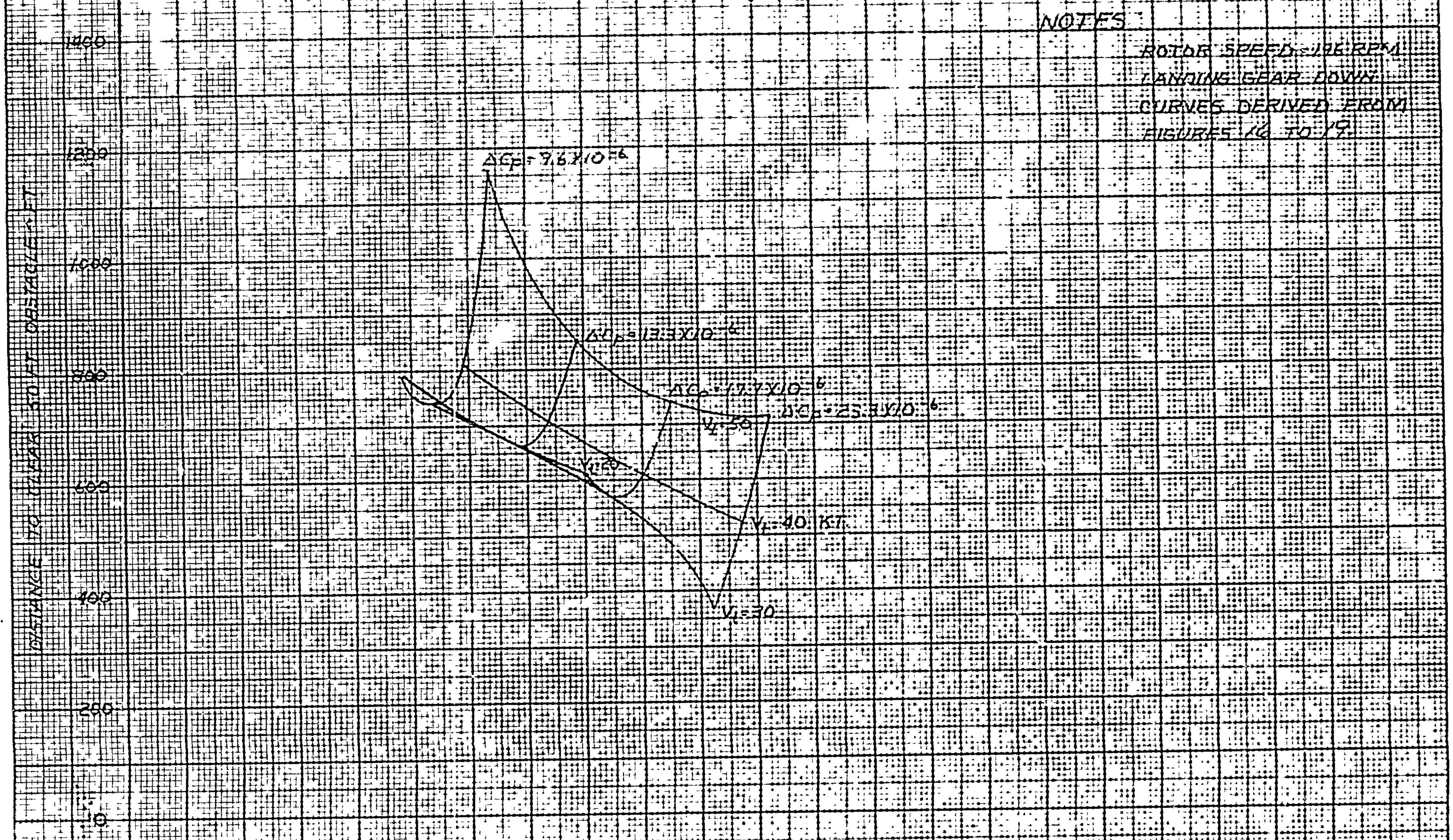

15-FT LEVEL ACCELERATION
Figure 18




15 FT LEVEL ACCELERATION

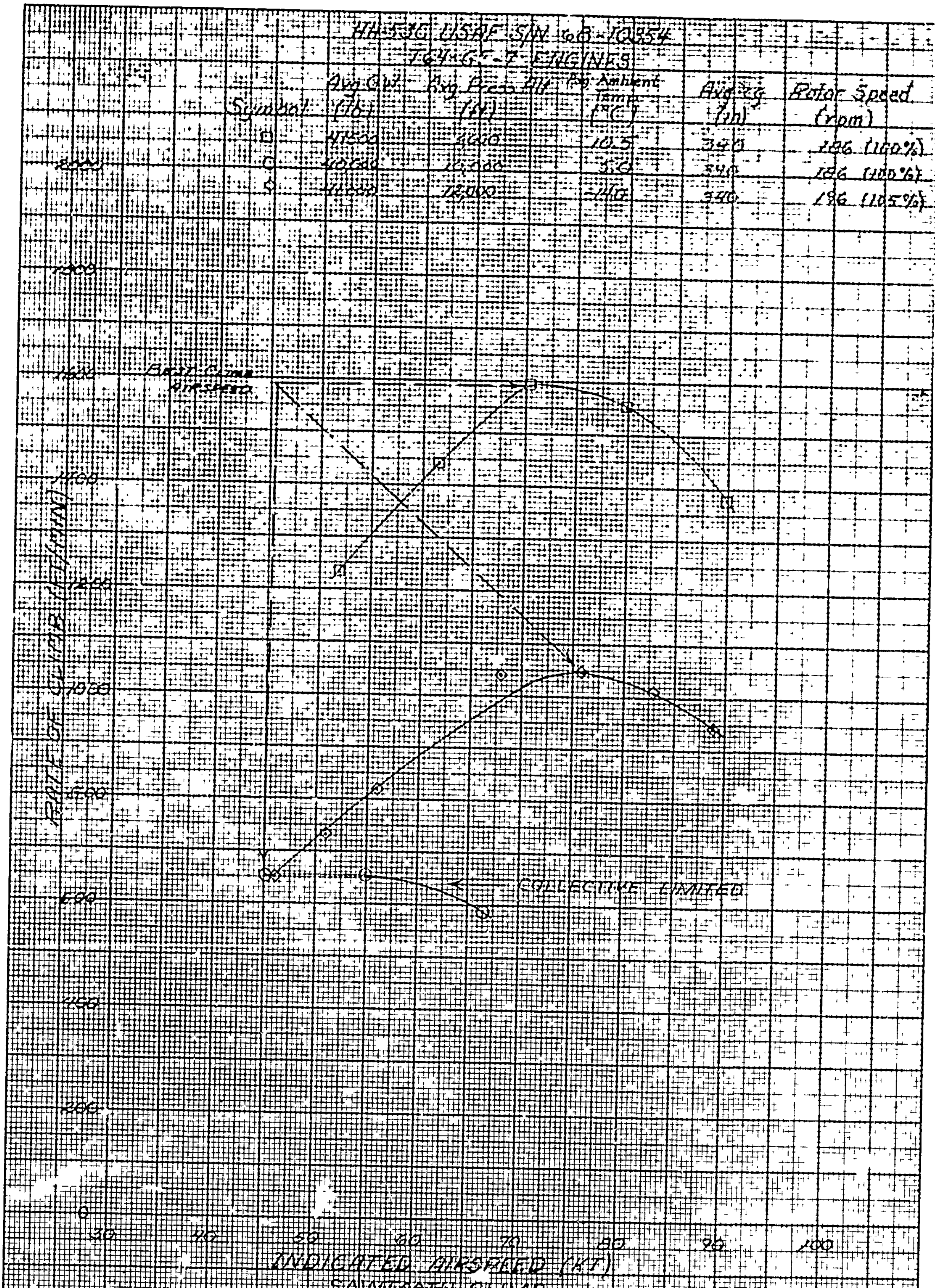
FIGURE 13

HH-53C USAF S/N 68-10354
T64-GE-7 ENGINES




15 FT. LEVEL ACCELERATION SUMMARY

Figure 20




SAWTOOTH CLIMB

Figure 21

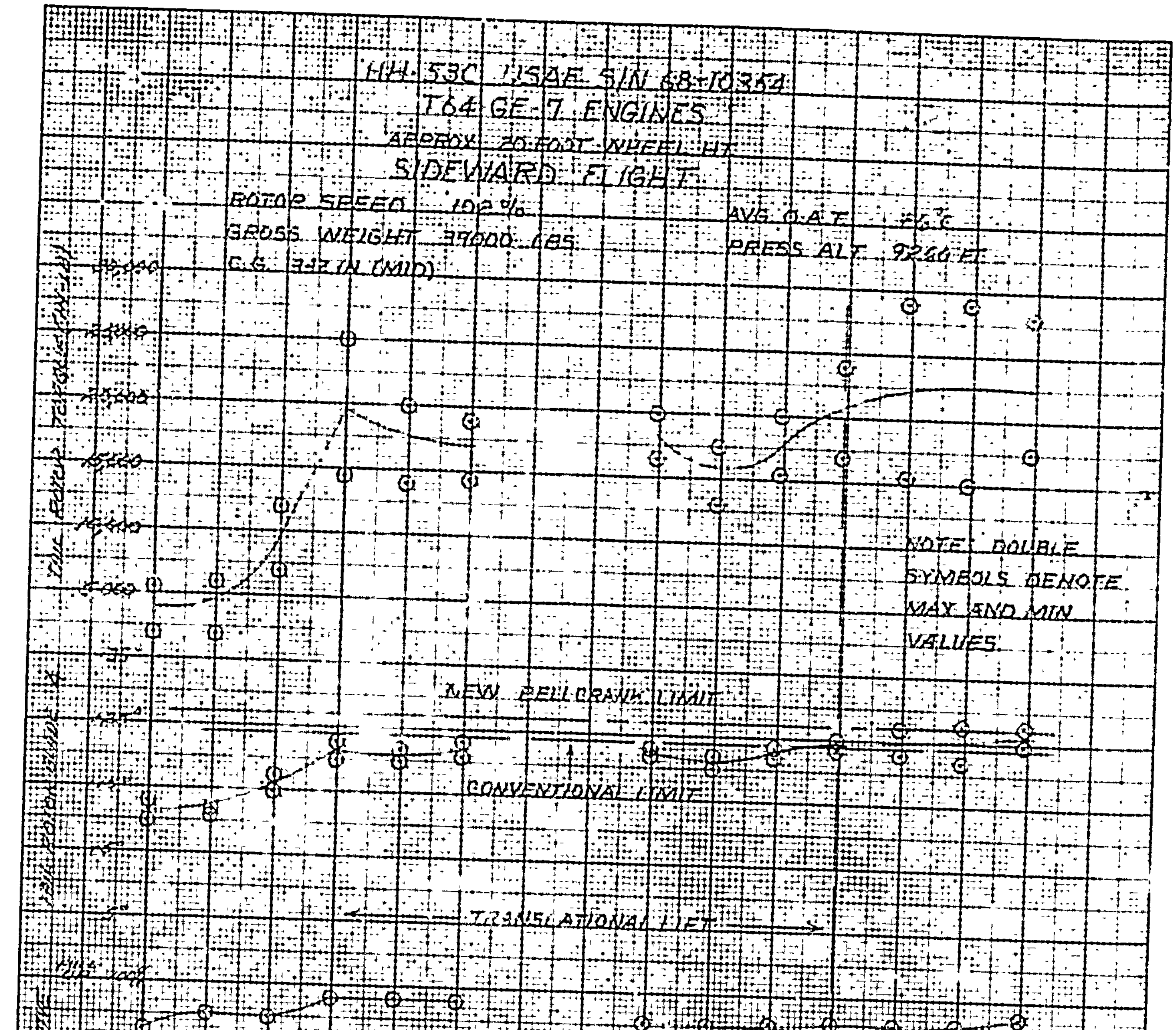





Figure 22 Sideward Flight

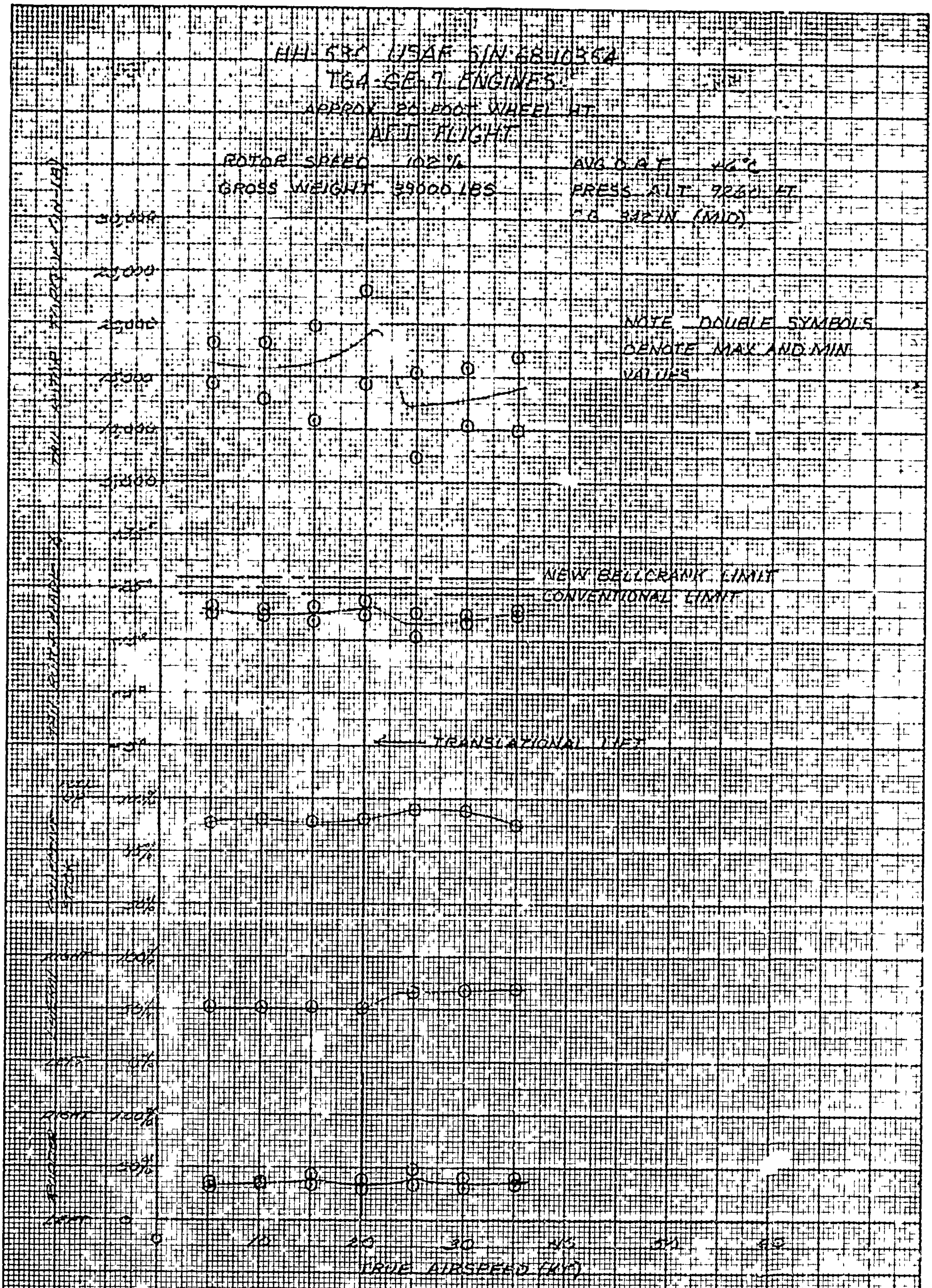



Figure 23 Paced IGE Flight

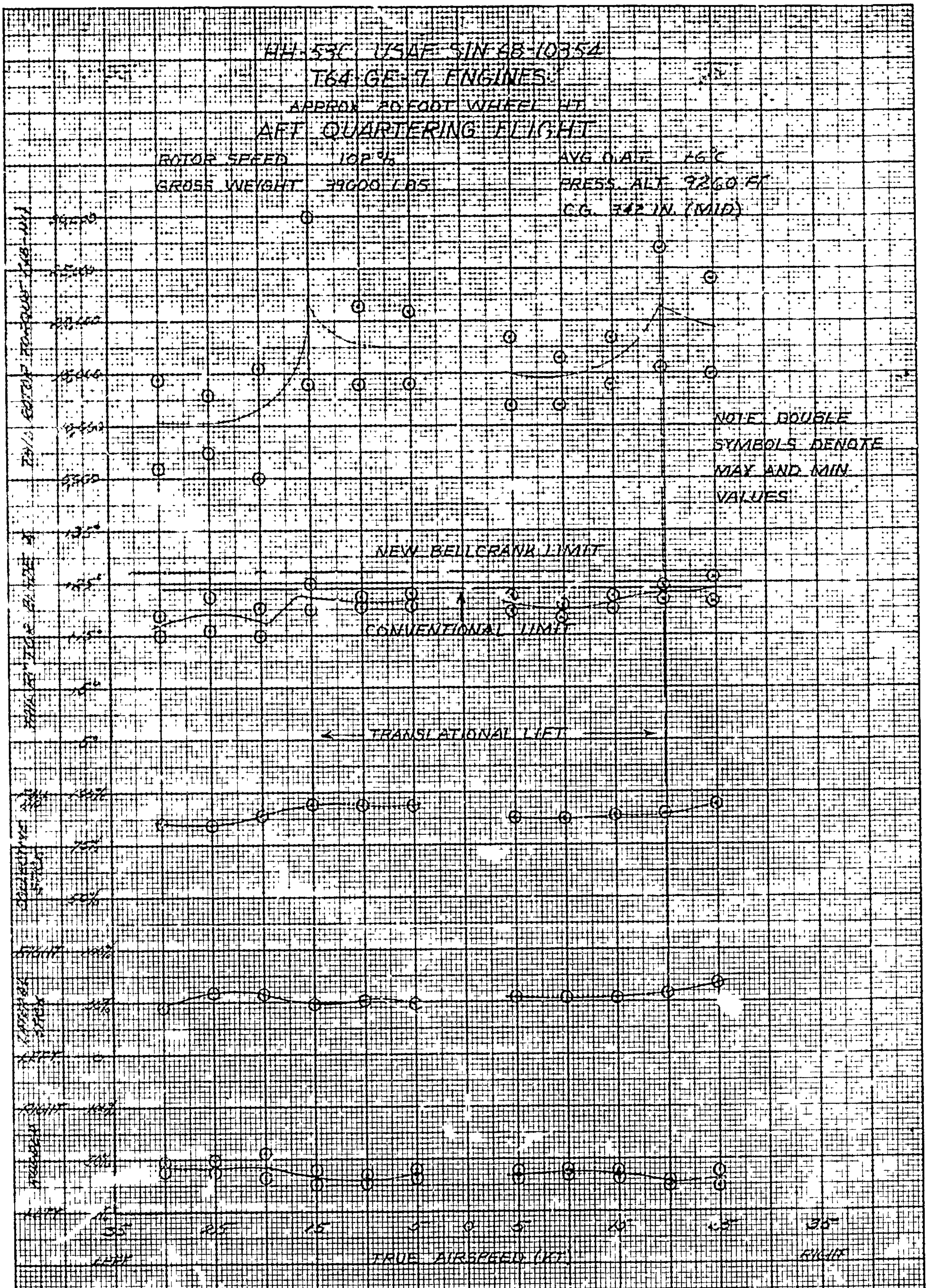



Figure 24 Paced IGE Flight

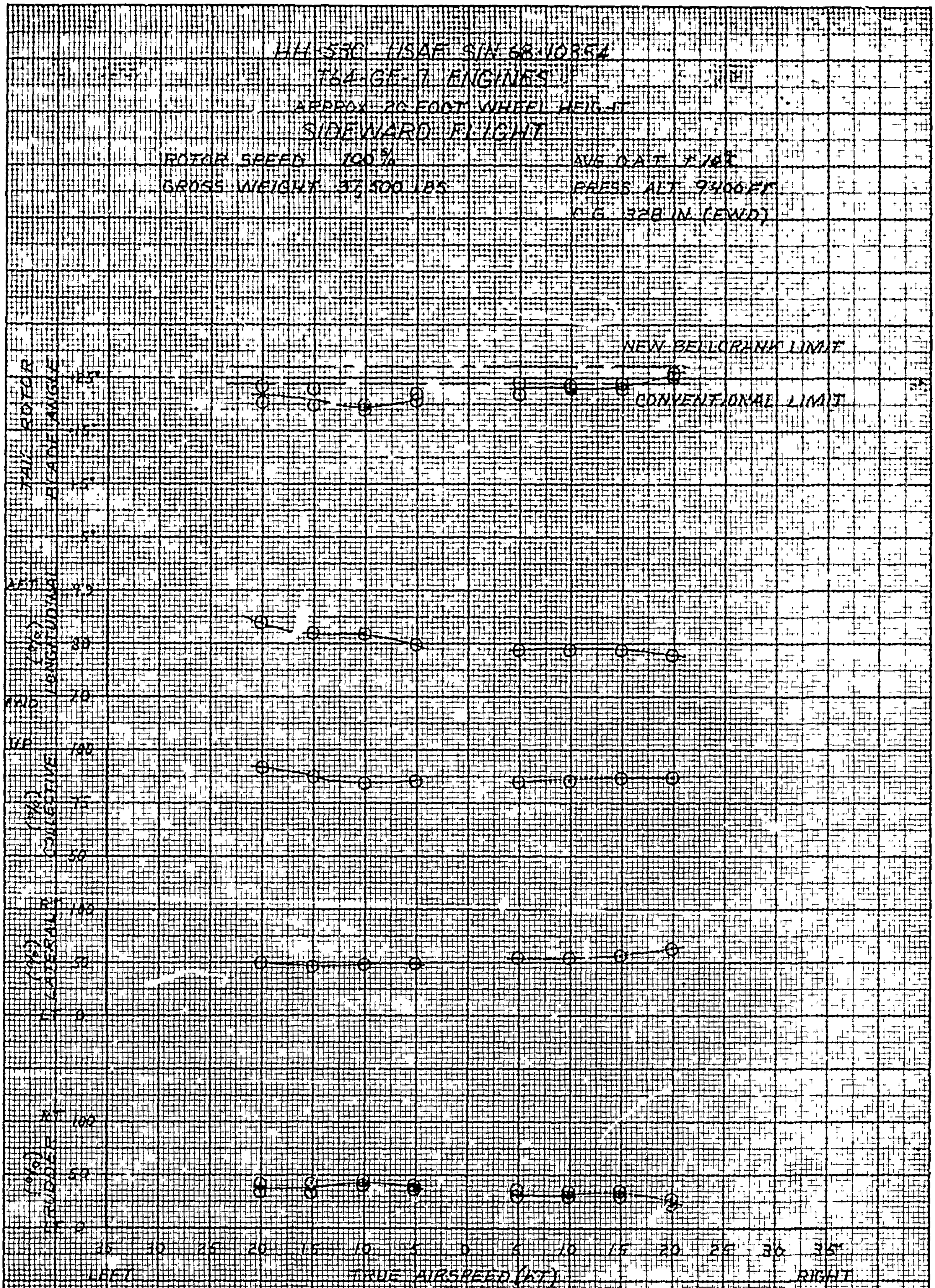
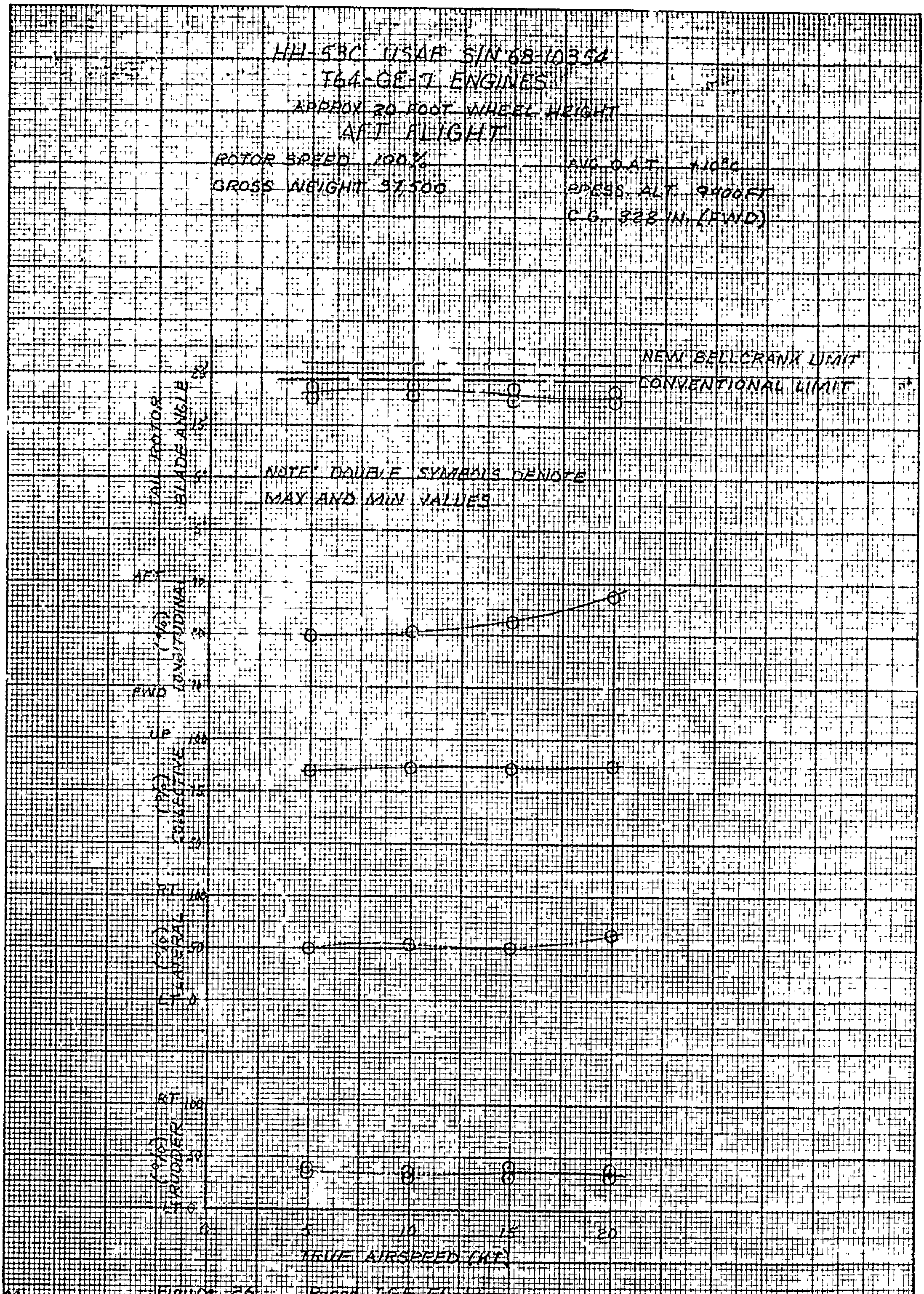



Figure 25 Paced IGE Flight




34 Figure 26 Paced IGE Flight

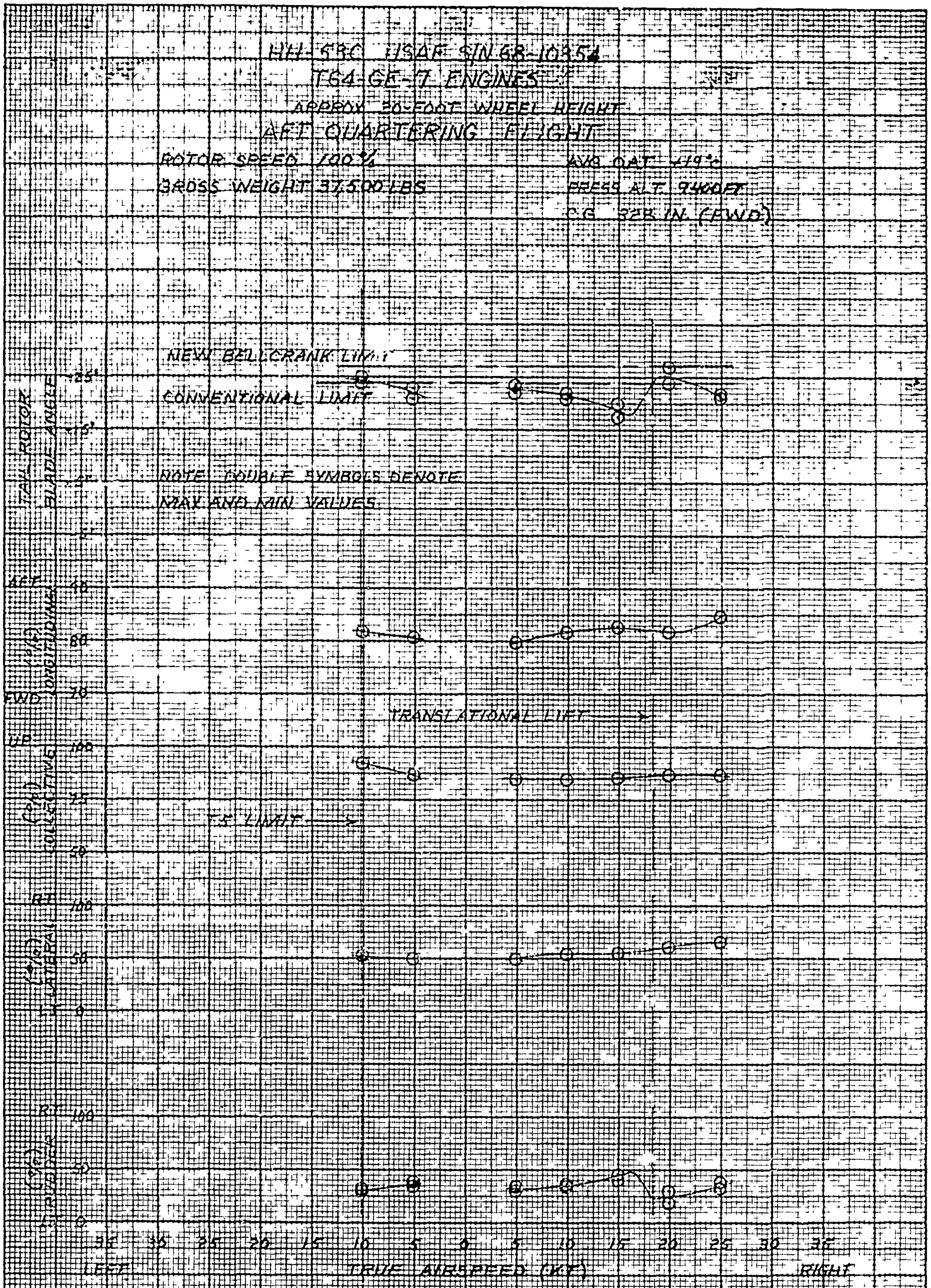



Figure 27 Faced AGL Flight

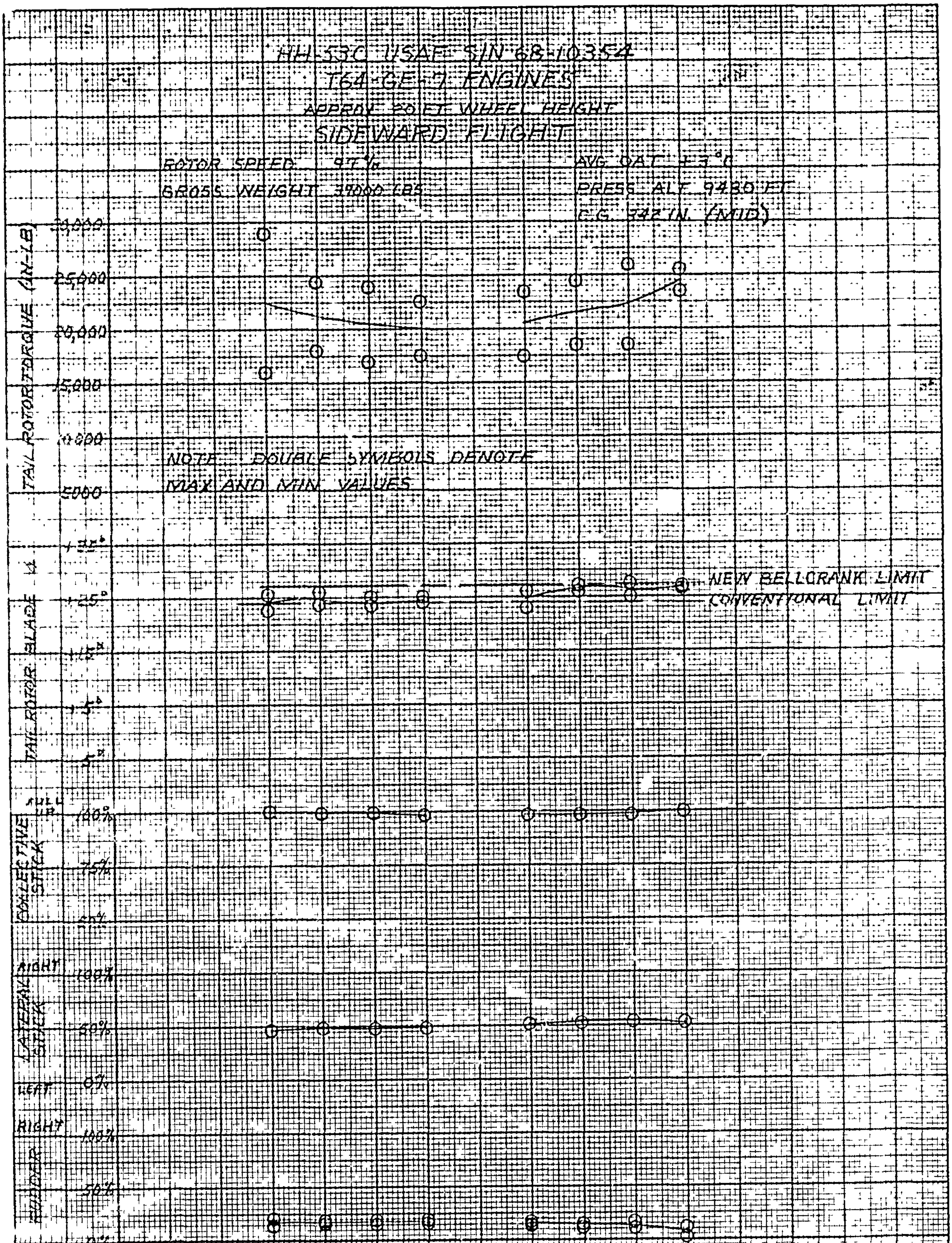




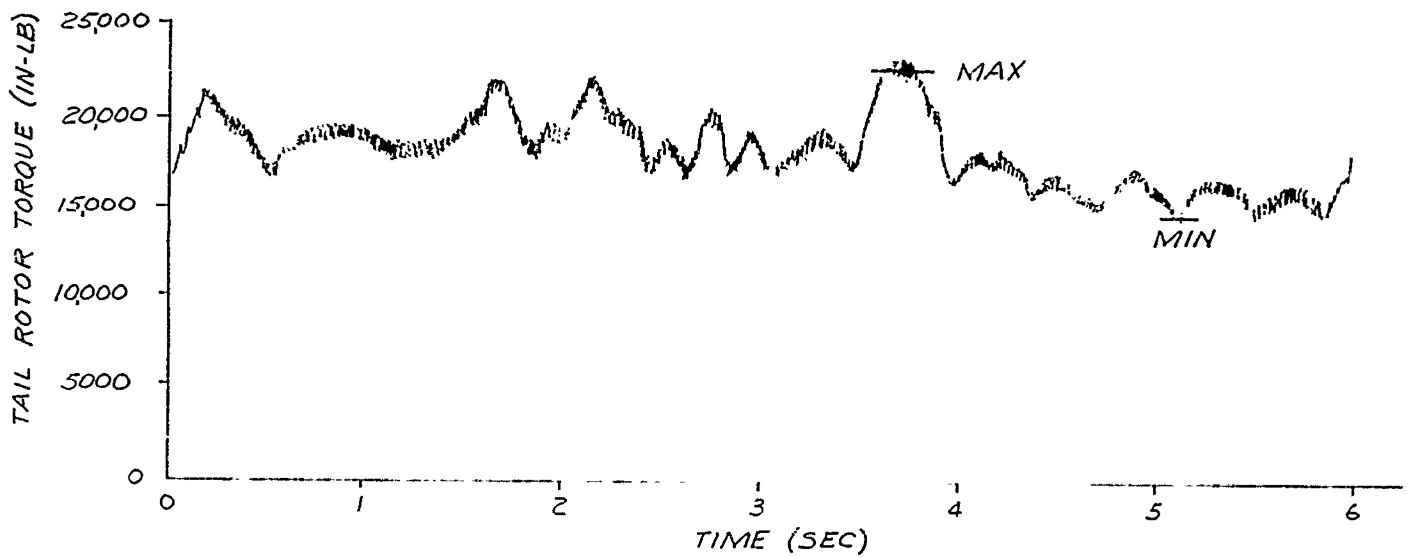

Figure 28 Paced IGE Flight

HH-53C USAF S/N 68-10354
T64-GE-7 ENGINES
AFT FLIGHT

ROTOR SPEED 102%
GROSS WEIGHT 39000 LBS

O.A.T. +6 °C
PRESS ALT 9260 FT
C.G. 342 IN (MID)

NOTE: TYPICAL TIME HISTORY FOR TAIL ROTOR TORQUE SHOWING MAX AND MIN VALUES AS PRESENTED IN FIGURES 22 THROUGH 28




TAIL ROTOR TORQUE TIME HISTORY

Figure 29

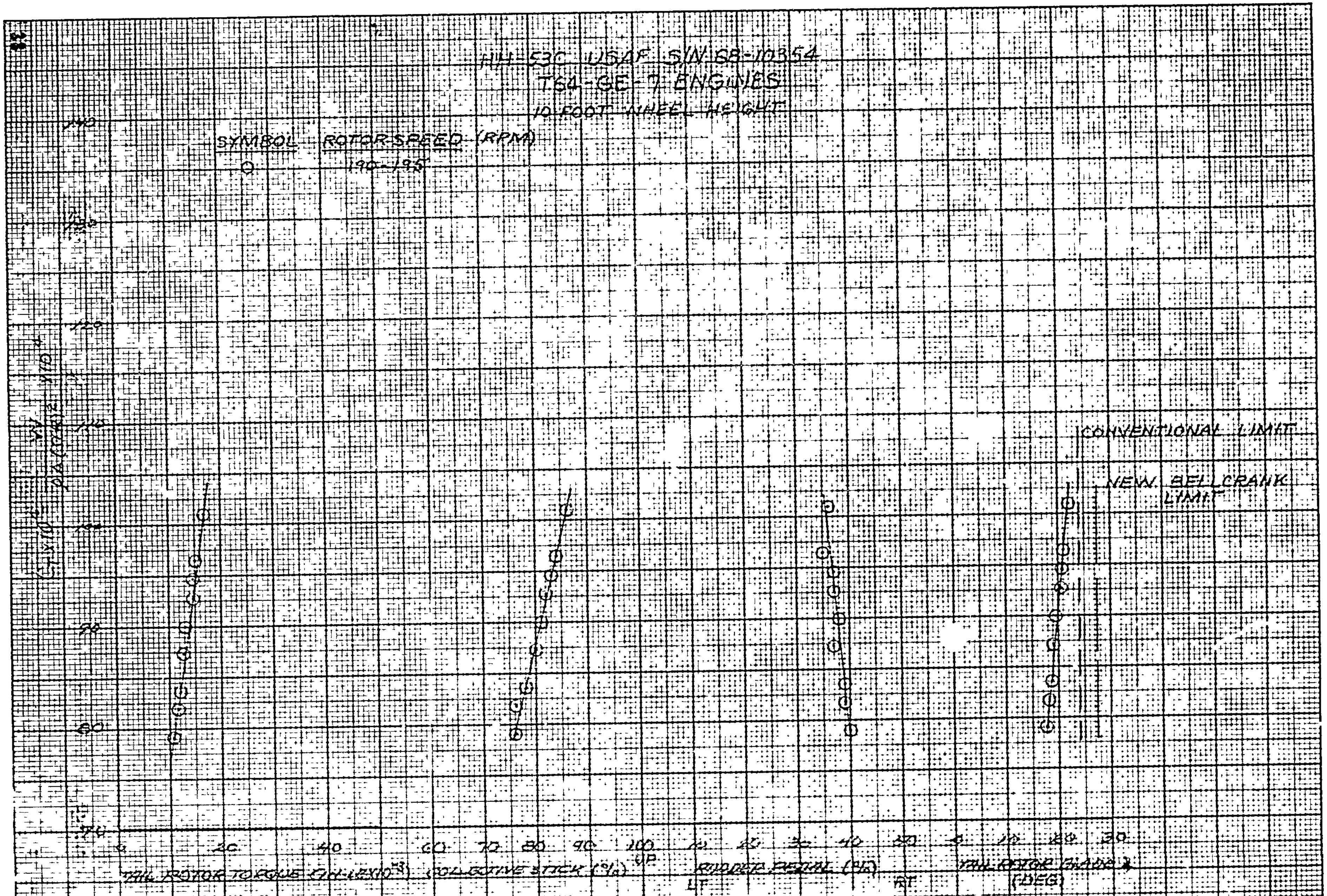



Figure 30 Hover Control Positions

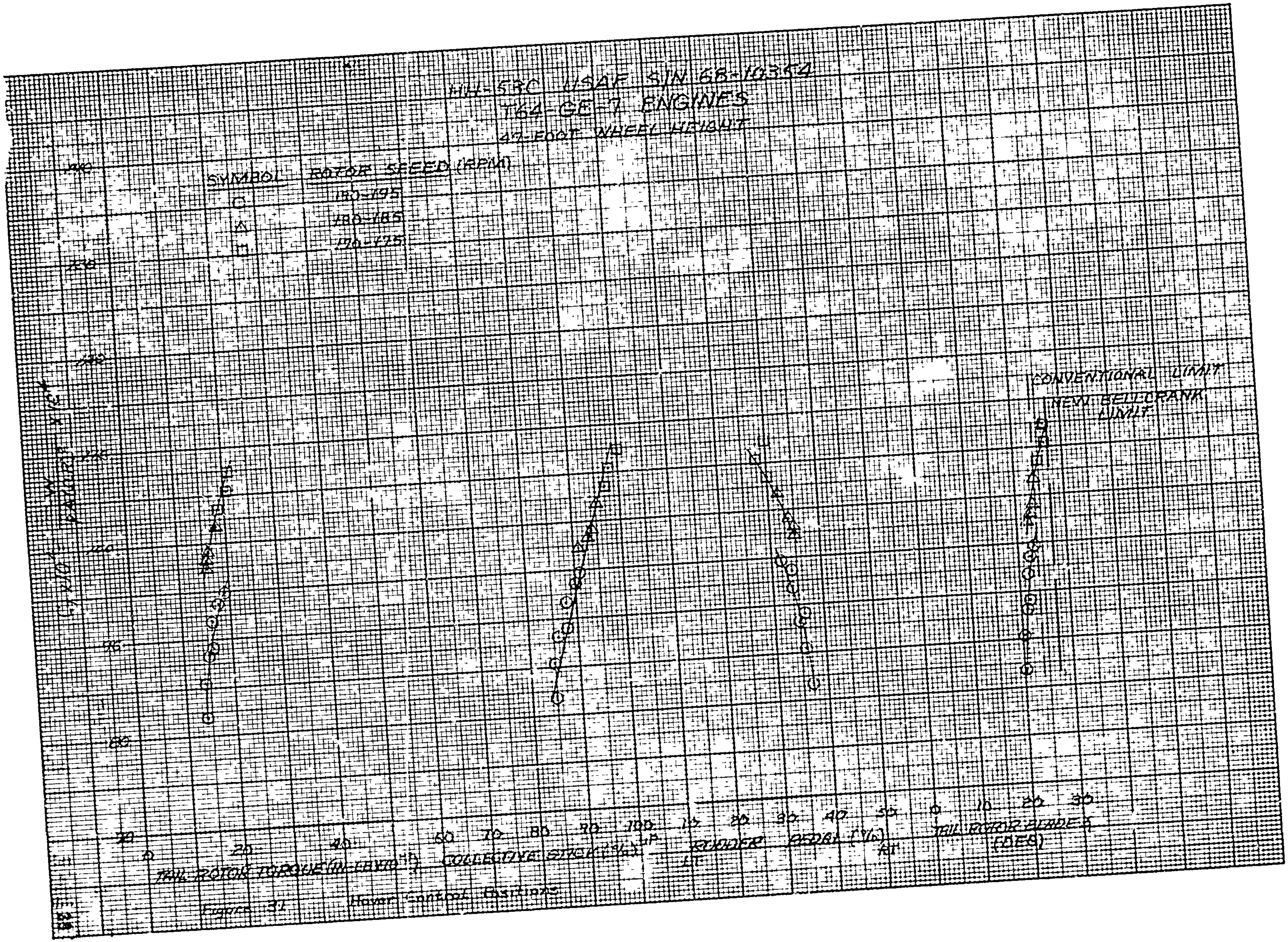



Figure 31 Hover Control Positions

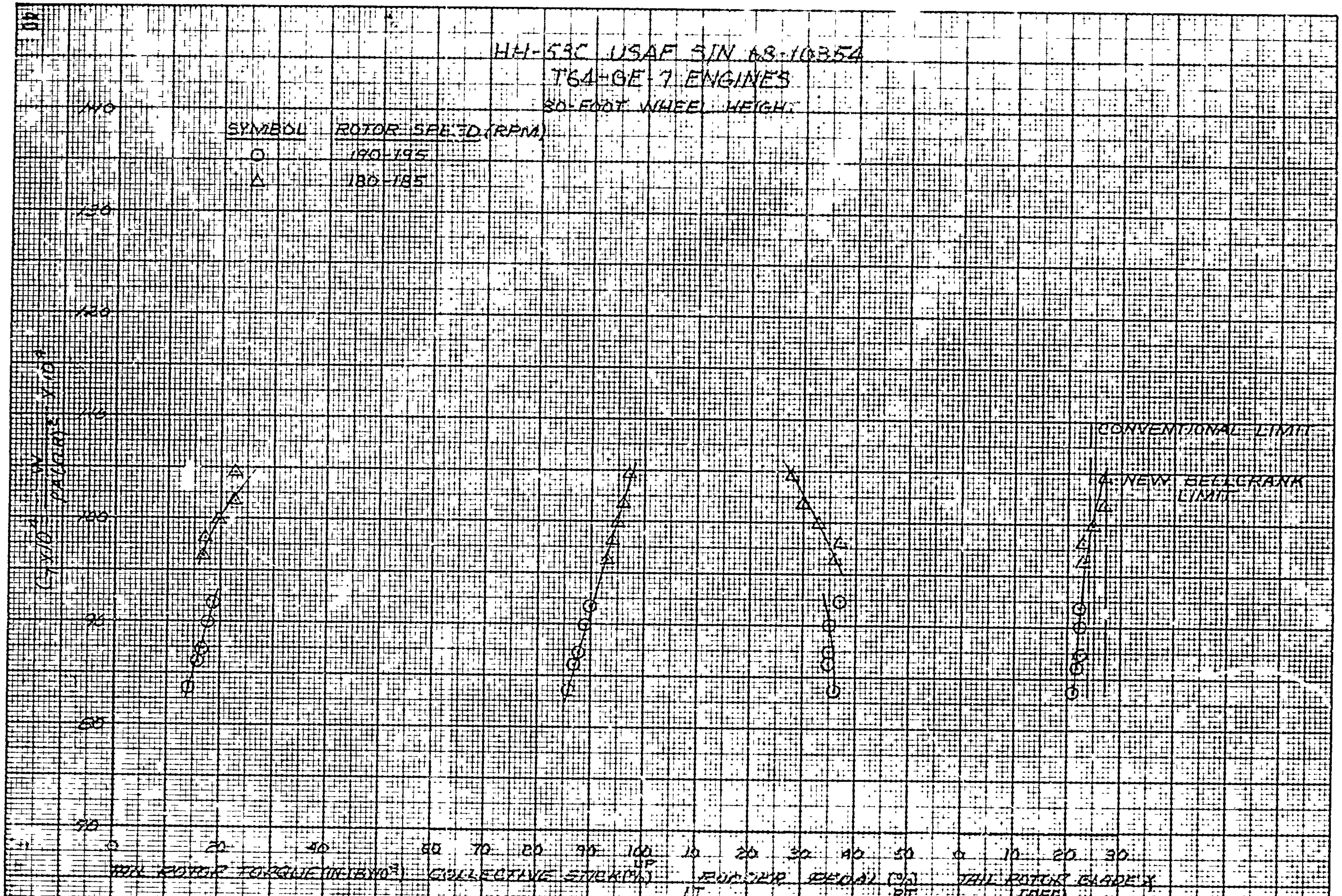



Figure 32 Hover Control Positions

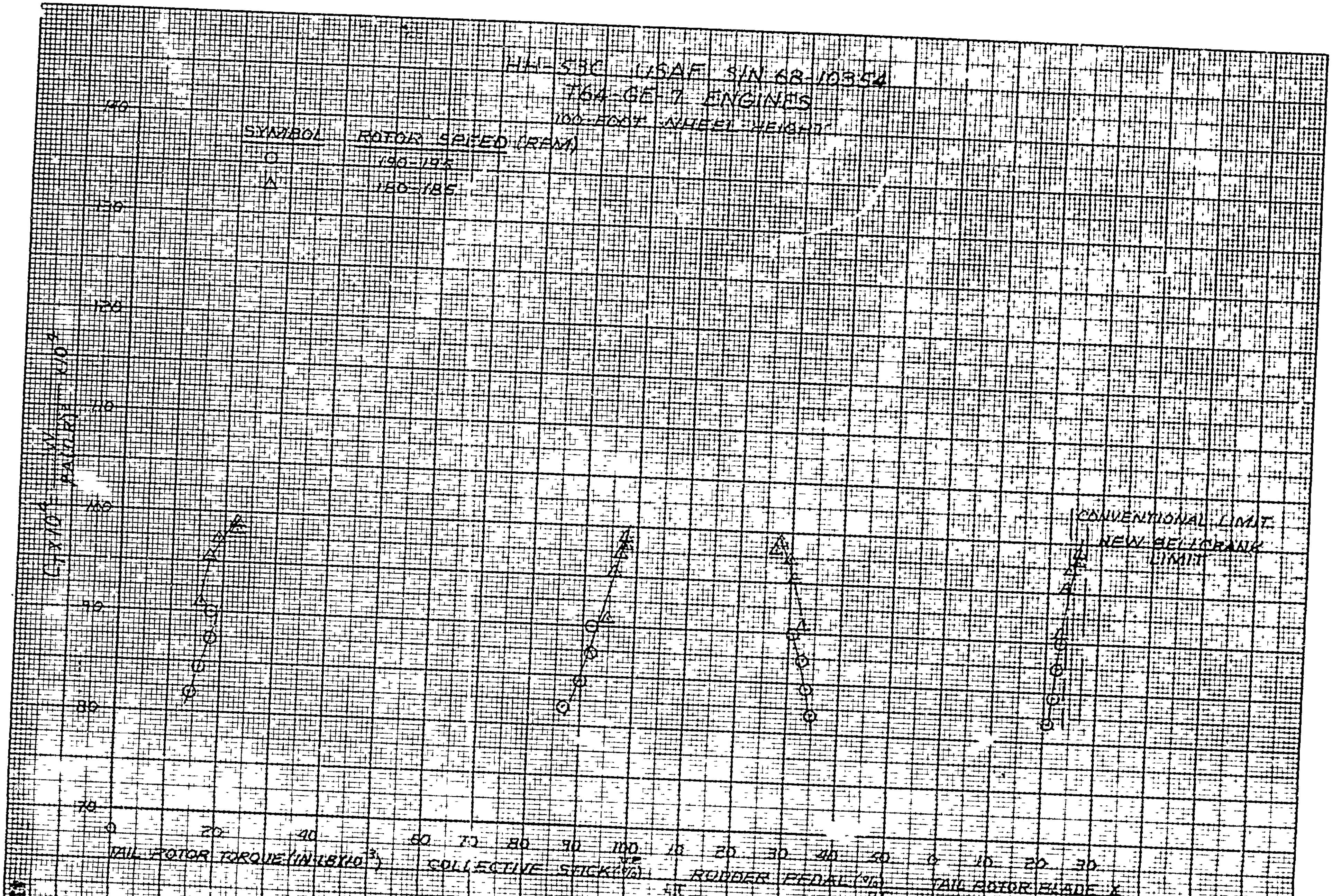
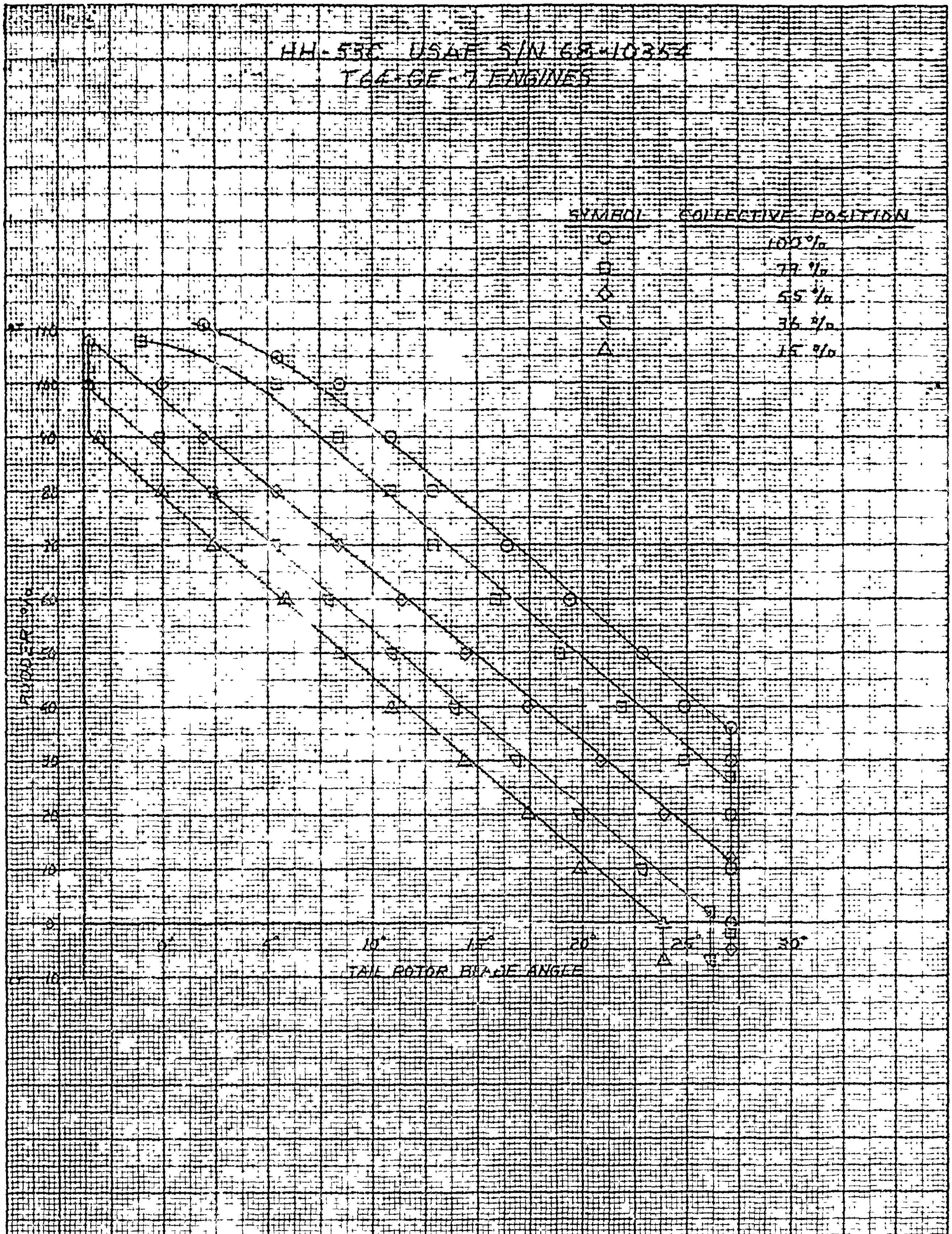
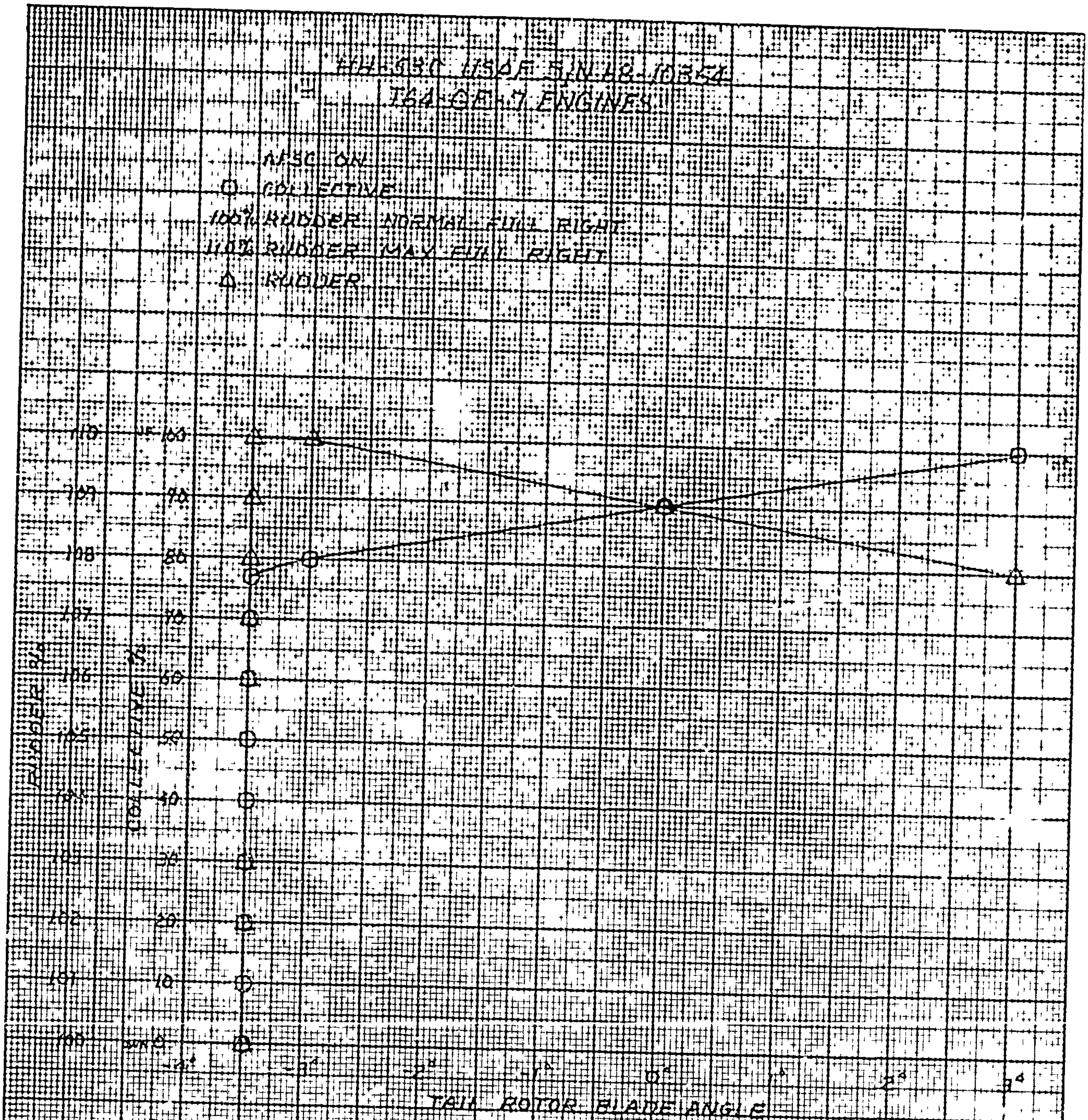



Figure 33 Hover Control Positions



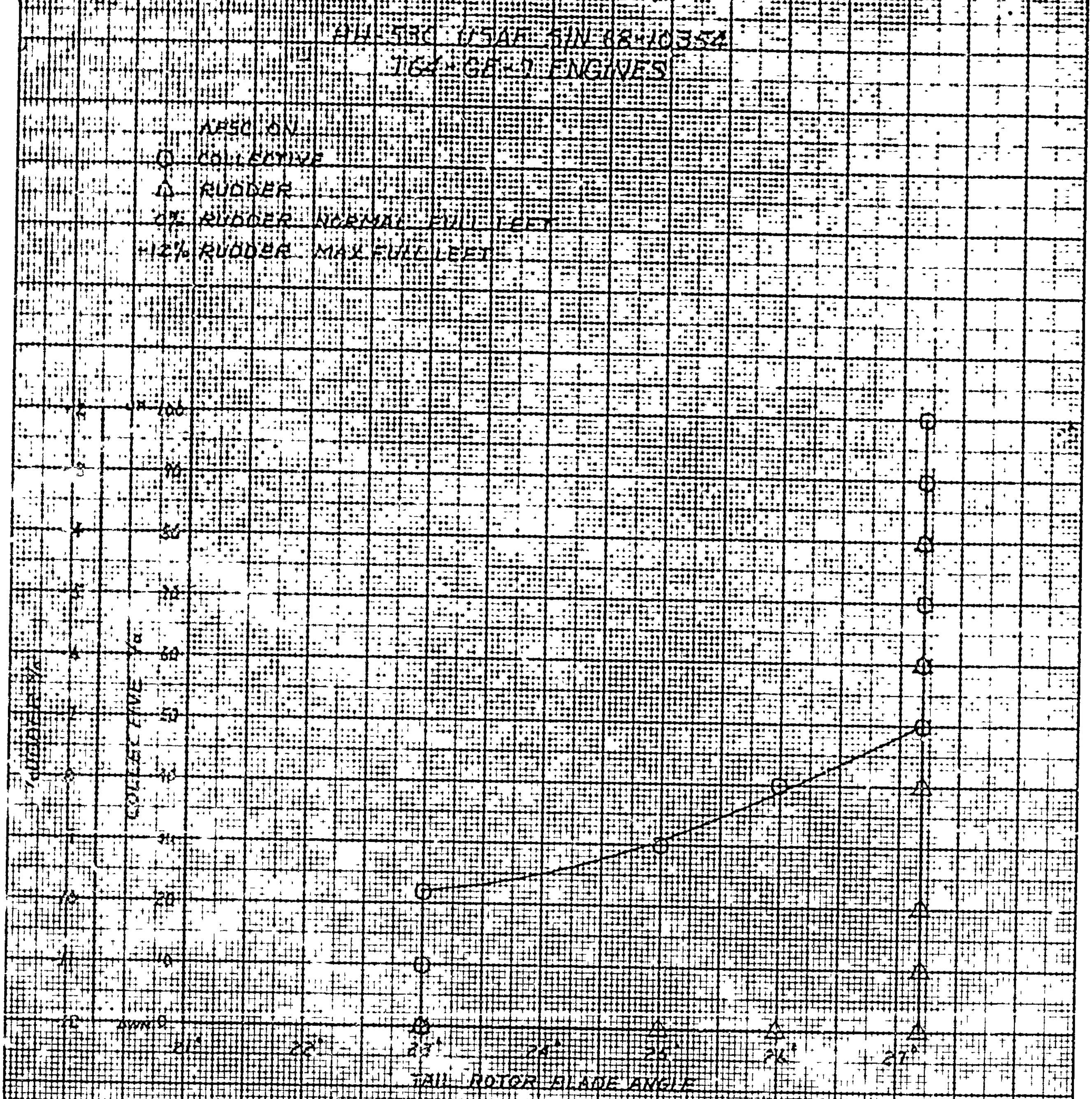

COLLECTIVE-RUDDER COUPLING

Figure 34




COLLECTIVE-RUDDER COUPLING

Figure 35




COLLECTIVE - RUDDER COUPLING

Figure 36

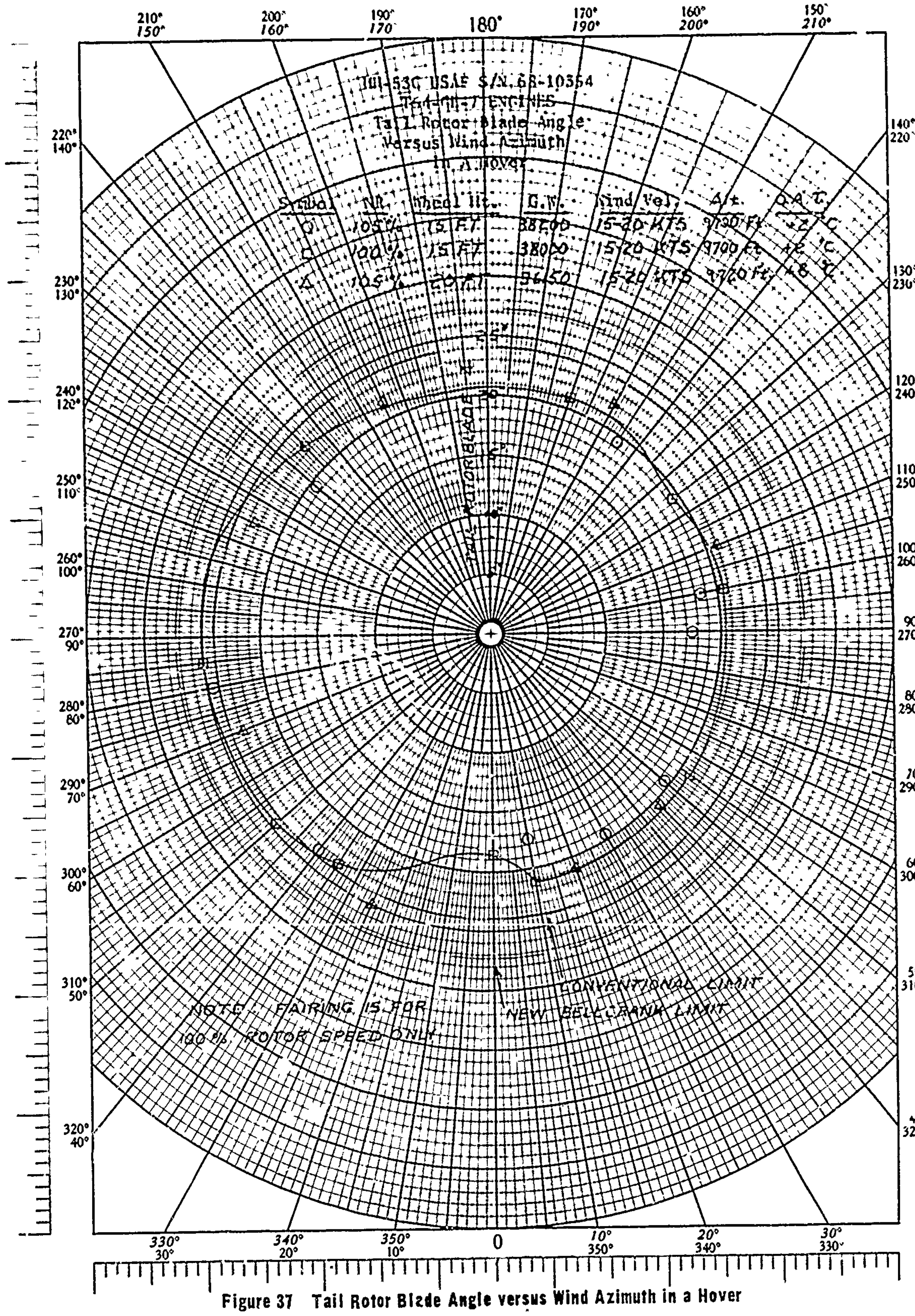



Figure 37 Tail Rotor Blade Angle versus Wind Azimuth in a Hover

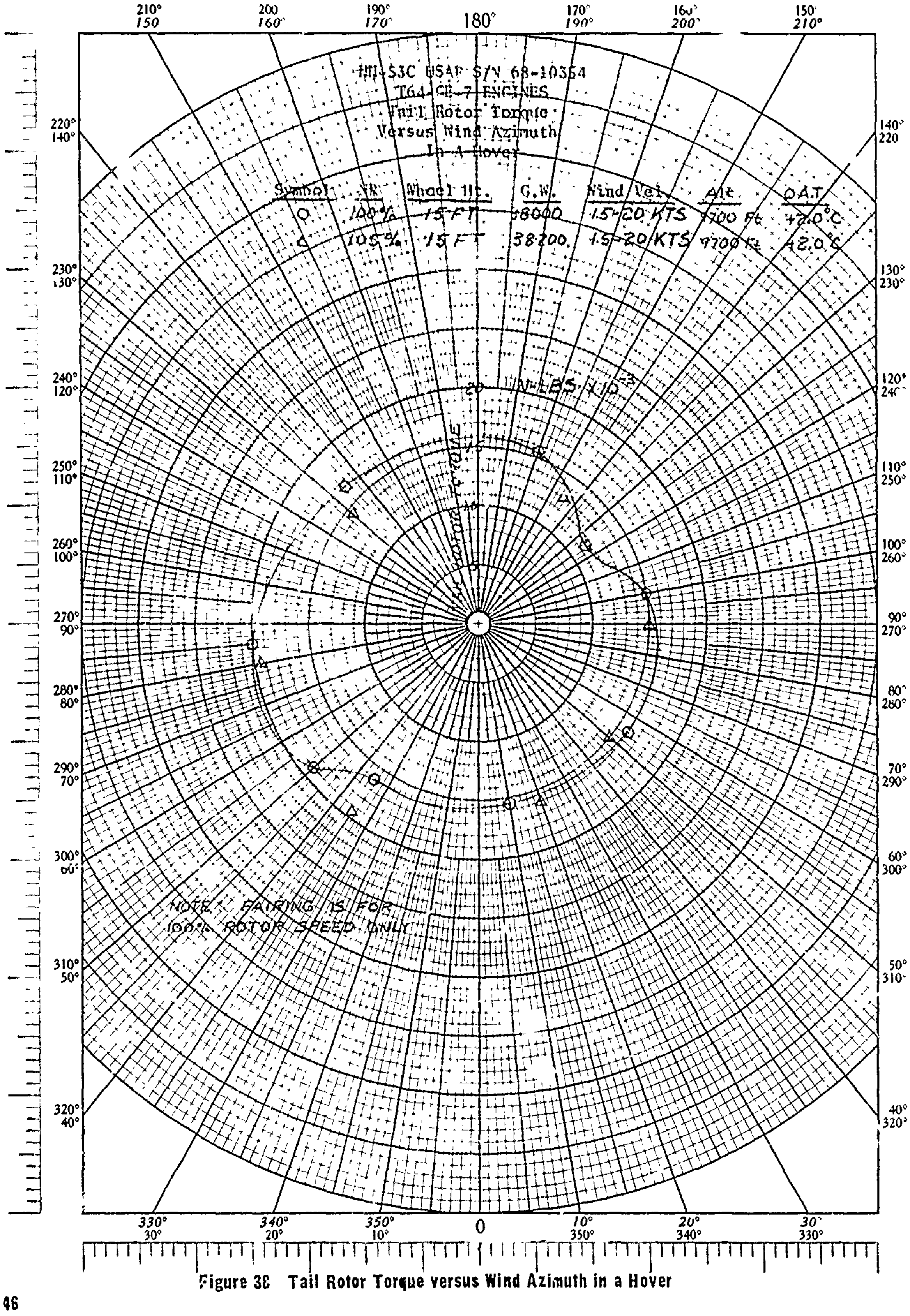



Figure 38 Tail Rotor Torque versus Wind Azimuth in a Hover

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2. Flight Manual USAF Series HH-53B, HH-53C Helicopter, T.O. 1H-53(H) B-1, 30 June 1970, changed 30 June 1971.

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13. ABSTRACT

This limited flight test program defined the hover and takeoff performance of the HH-53C helicopter at high altitude and included an evaluation of the increased tail rotor power provided by a new tail rotor bellcrank. Hover and takeoff performance were satisfactory during high altitude testing. Operating at 105-percent rotor speed at high altitude, heavy gross weight combinations provided a more rapid response of the aircraft to control inputs than operation at 100-percent where control response was somewhat sluggish. In addition, it provided increased performance, prevented reaching mechanical control stops from limiting performance and lowered cruise guide readings. The new tail rotor bellcrank increased the HH-53C lift capability approximately 4,500 pounds by preventing tail rotor authority from limiting aircraft performance. A compromise between maximum performance and safety resulted in recommendation of a 35 knot indicated airspeed for takeoff.

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